ariane

user's manual

1980 Issue

ariane
user's manual
1980 Issue

Approved by

R. ORYE

Head of Ariane Department ESA

R. VIGNELLES

Head of Ariane Project CNES/AR

esa
european space agency
agence spatiale européenne

cnes
centre
national
d'études
spatiales

Publication notice to the holders of the Ariane user's manual

Revisions to this document will be issued-periodically to reflect changes and aditions to vehicle performance, operations and spacecraft design restraints. Unless the recipient is on the distribution list for revision sheets, his manual will very soon become outdated. You are therefore encouraged to return the reader's service card promptly.

Also if you have any comments or suggestions concerning content or format, please note on the service card.

Inquiries concerning clarification or interpretation of the material contained herein should be directed to the :

Ariane Department
European Space Agency

8-10, rue Mario-Nikis
75738 Paris Cedex 15

Ariane user's manual

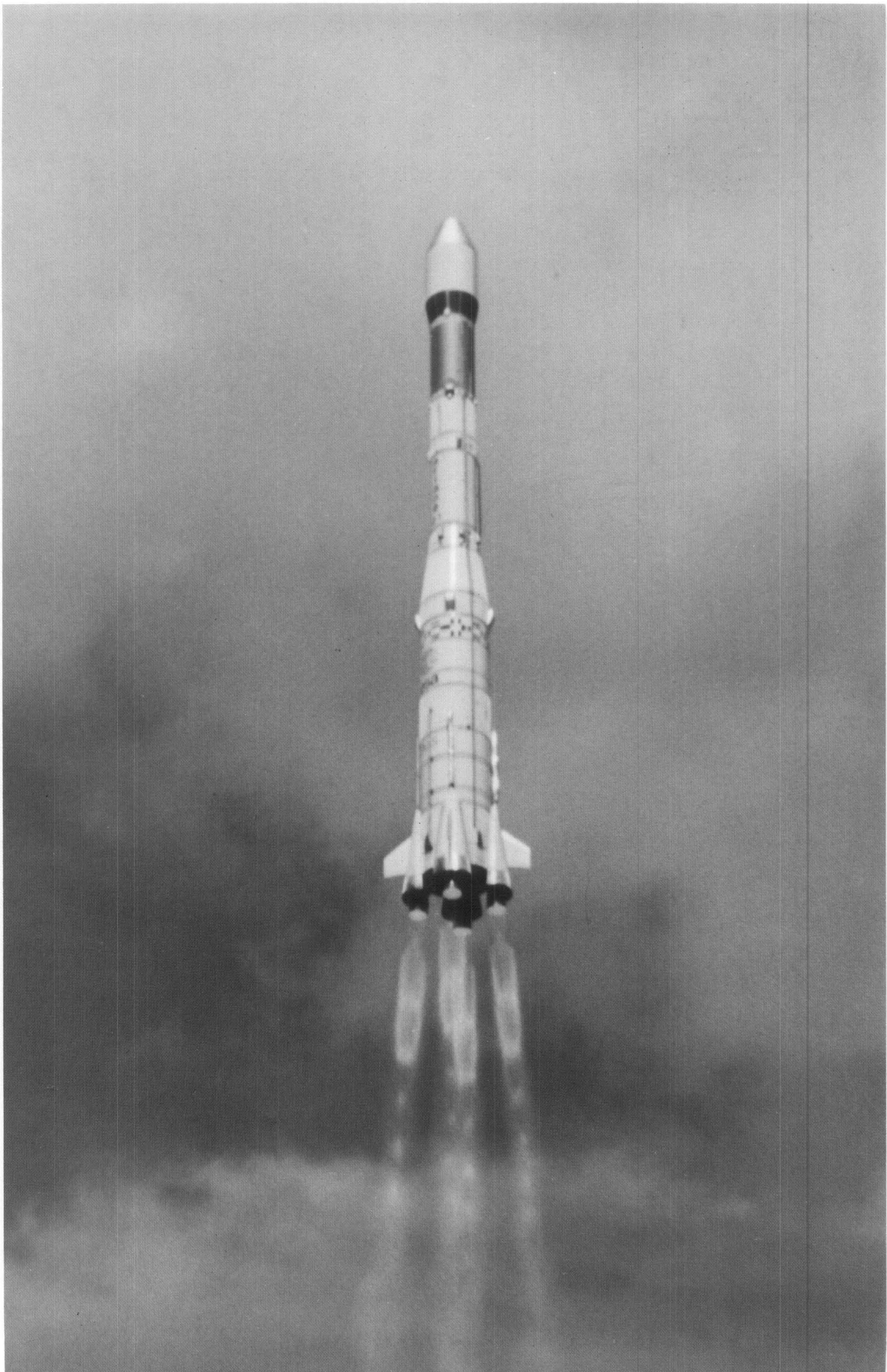
Serial Nr.

1980 Issue

If you wish to receive future revisions to the manual please fill out and return this card.

☐ Note change of address :

Address :	Name :
	Date :
Comments/Suggestions :	



ARIANE L 01 vehicle after lift-off

Contents

List of figures
Ariane user's manual

Chapter 1 / Introduction

Chapter 2 / Performance data

Table of contents
Ariane user's manual

Chapter 1 / Introduction

Chapter 2 / General performance data

List of figures

Chapter 3 / launch vehicle / payload interface

Table of contents

Chapter 3 / Launch vehicle / payload interface

List of figures

chapter 3

Chapter 4 / Launch operation

Table of contents

chapter 3

Chapter 4 / Launch operation

List of figures

Chapter 4

Table of contents

Chapter 4

List of figures

Chapter 5 / Safety

Table of contents

Chapter 4

Chapter 5 / Safety

List of figures

Chapter 6 / Documentation

Chapter 7 / General configuration of Sylda beneath the fairing

Table of contents

Chapter 5

Chapter 6 / Documentation

Chapter 7 / Dual launch system with Sylda

List of figures

Chapter 7

Table of contents

Chapter 7

List of figures

Annex 1 / Description of launch vehicle

Table of contents

Chapter 7

List of figures

Annex 2 / Guiana Space centre

Table of contents

Annex 2 / Guiana Space Centre

Annex 3 / Abbreviations

Introduction — Chapter 1

1.1. Purpose of the user's manual

This manual is designed to provide Ariane users with the information they need in order to prepare payload projects.

The reader should also refer to the " Safety Regulations " and " Guiana Space Centre (CSG) Manual ".

These three documents constitute the technical reference documentation for the Ariane launch system, used for the Ariane-payload feasibility study.

On completion of the feasibility phase, formal documentation will be established in accordance with the instructions given in chapter 6 of this manual.

1.2. General organization

The general organization of the activities associated with the launch of a payload by Ariane is based on the principle of a dialogue between a payload officer and an Ariane officer, to be appointed by their respective authorities not later than the date at which the launch contract is signed. For a given launch, therefore, there are :

- one or two payload officers, depending on whether the launch is a single or a dual one ;
- an Ariane officer.

Figure 1.1. shows the principle of the organization throughout the activities associated with the launch contract.

Chapter 4, paragraph 5, details the operational organization of a launch in which the duties of the Assistant Mission Head (" Adjoint au Chef de Mission " - ACM) are carried out by the Ariane officer, thus maintaining the principle of Ariane being represented by a single interlocutor vis-à-vis the payload authority.

Chapter 6 lays down the documentation required for preparing for the launch of a payload by Ariane.

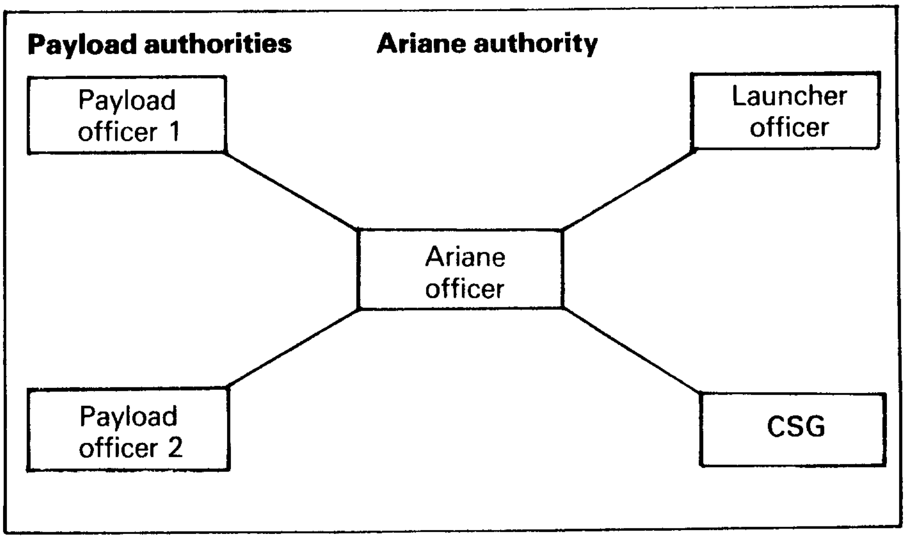


Fig. 1.1. Principle of user / Ariane relationship

1.3. Main characteristics of the Ariane launch vehicle

Ariane is a three-stage launch vehicle using liquid propellants, with an inertial guidance system.

The 1st and 2nd stages burn UDMH and N_2O_4, the 3rd stage being of the cryogenic type (liquid hydrogen and liquid oxygen). Ariane has a height of approximately 47 m, and a lift-off mass of some 210 tonnes.

A brief description of the vehicle is given in Annex 1.

Figure 1.2 shows the system of reference axes, and figure 1.3 a cut-away view of the launch vehicle.

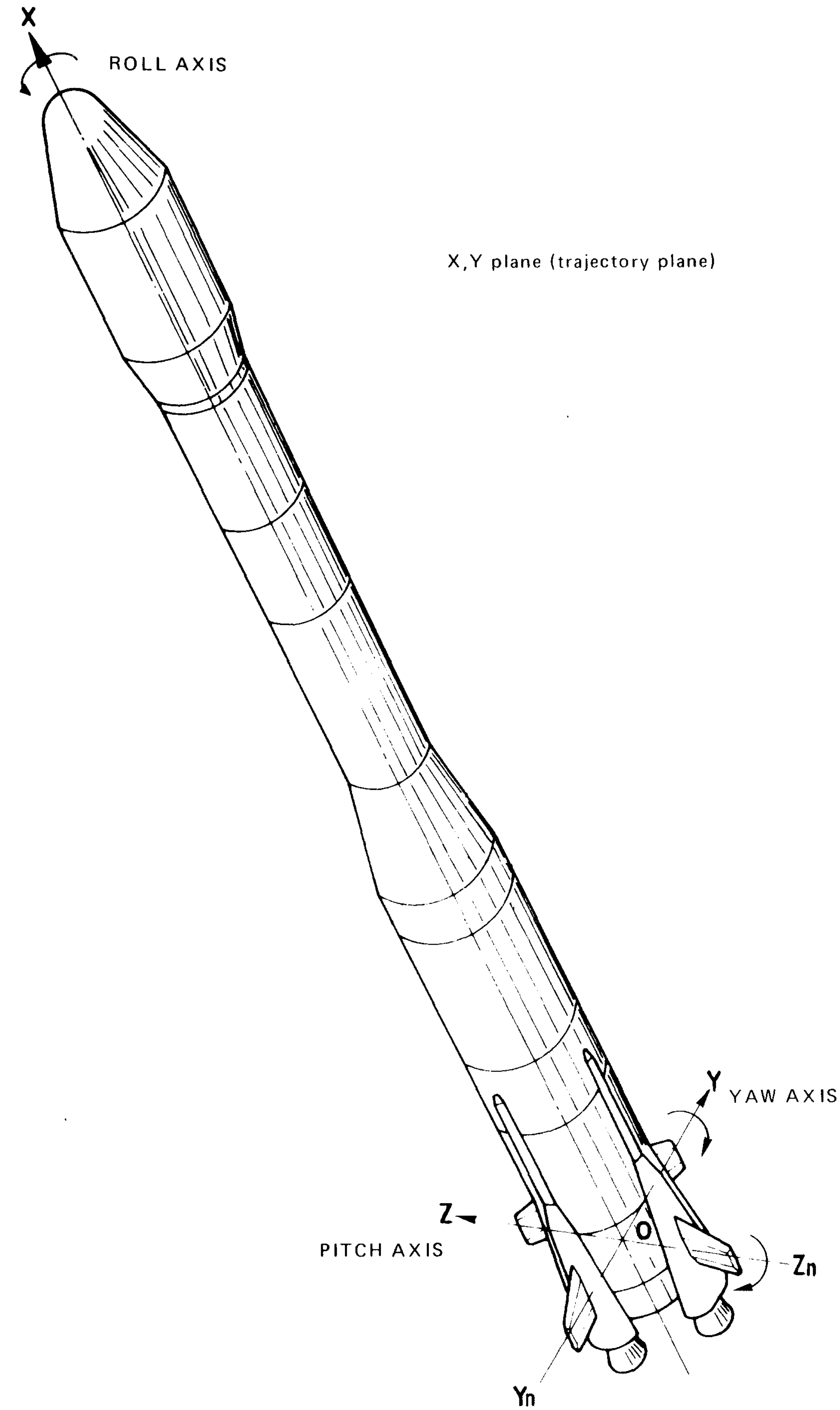


FIG. 1.2 LAUNCH VEHICLE AXES

1. Typical payload
2. Fairing separation plane
3. 3rd stage / payload separation plane
4. Fairing
5. Equipment bay subsystems
6. Payload adaptor
7. Vehicle equipment bay (VEB)
8. Tightness membrane
9. Antennas
10. LH tank
11. Anti-sloshing
12. LOX tank
13. Acceleration rockets (4)
14. Roll and attitude command system (RACS)
15. 2/3 separation plane
16. 3rd stage thrust frame
17. 3rd stage engine HM7
18. Helium pressurization tank
19. Helium pressurization tanks
20. Interstage 2/3
21. 2nd stage front skirt
22. Retro rockets (3)
23. Anti-sloshing
24. N2O4 tank
25. UDMH tank

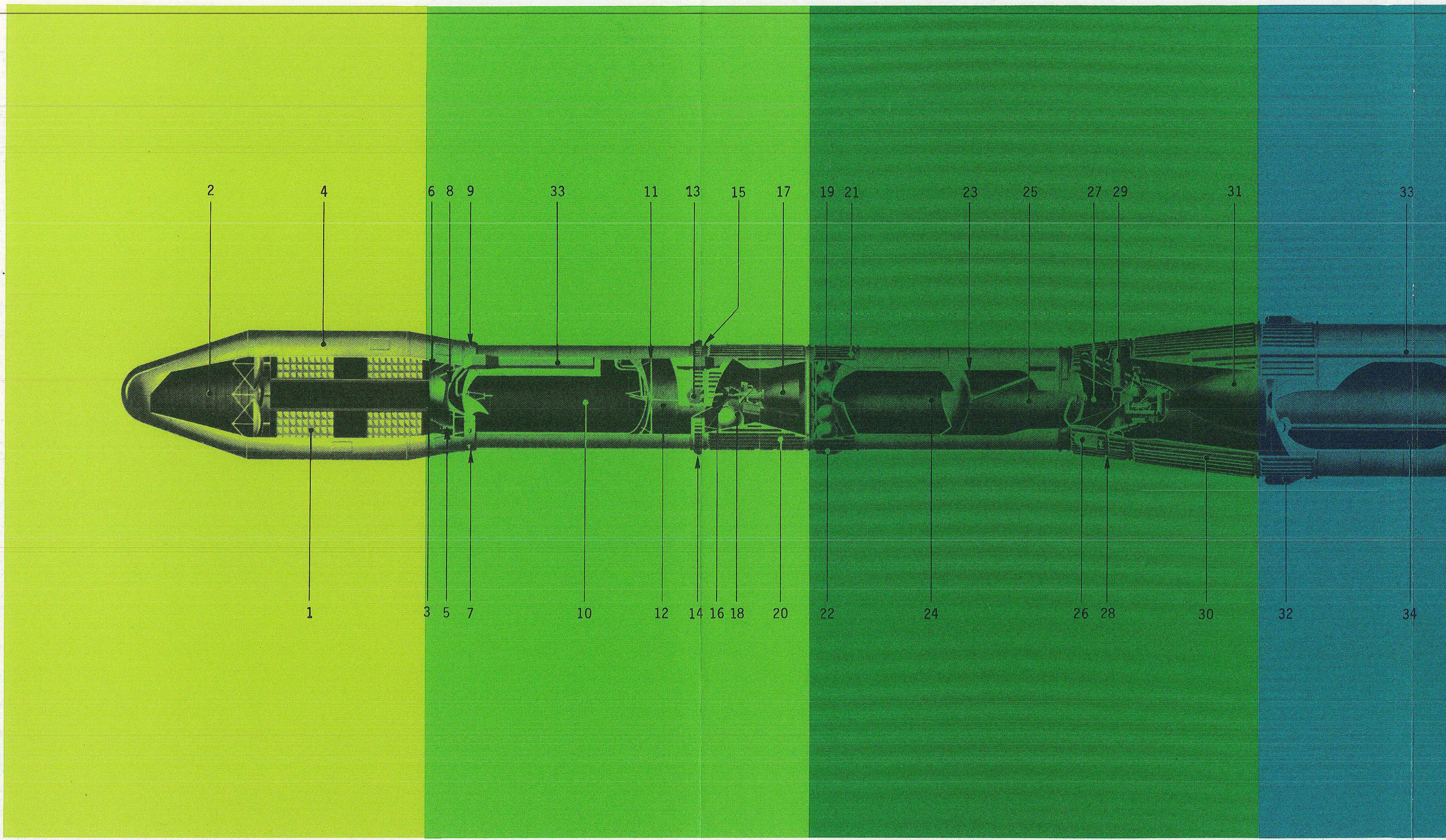


FIG. 1.3 LAUNCH VEHICLE INBOARD PROFILE

Edit

tion plane

ns

VEB)

9. Antennas
10. LH tank
11. Anti-sloshing
12. LOX tank
13. Acceleration rockets (4)
14. Roll and attitude command system (RACS)
15. 2/3 separation plane
16. 3rd stage thrust frame
17. 3rd stage engine HM7
18. Helium pressurization tank
19. Helium pressurization tanks
20. Interstage 2/3
21. 2nd stage front skirt
22. Retro rockets (3)
23. Anti-sloshing
24. N2O4 tank
25. UDMH tank
26. Acceleration rockets (6)
27. 2nd stage thrust frame
28. 1/2 separation plan
29. 2nd stage toric water tank
30. Interstage 1/2
31. 2nd stage engine — 1 Viking IV —
32. Retro rockets (8)
33. External cable duct
34. N2O4 tank
35. N2O4 pipes
36. Intertank skirt
37. UDMH tank
38. 1st stage thrust frame
39. 1st stage toric water tank
40. Fins
41. Fairings
42. 1st stage engines — 4 Viking V —

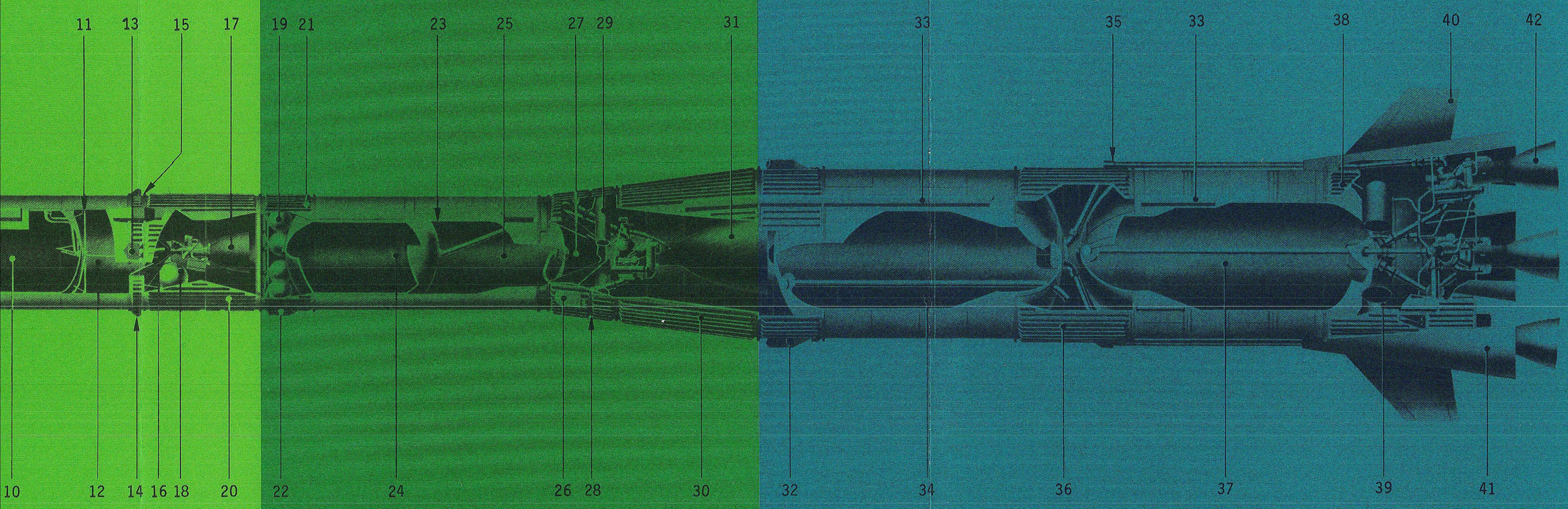


FIG. 1.3 LAUNCH VEHICLE INBOARD PROFILE

General performance data — Chapter 2

2.1. Introduction

This chapter presents Ariane performance data, so as to provide the user with the maximum payload-mass value for the purposes of his mission.

Among the many possible missions, the following four categories may be singled out :

- the launching into transfer orbit of geostationary satellites, this being the basic Ariane programme objective.
- the launching of Earth satellites into elliptical orbit.
- the launching of Earth satellites into sun-synchronous orbit.
- the launching of space probes outside the Earth's field of gravity.

2.2. Definitions

2.2.1 Launch vehicle mission

The Ariane launch vehicle has fulfilled its mission when it has delivered the payload to the selected orbit, with the intended attitude and within the limits of the specified dispersions.

Ariane can also spin the payload, up to a maximum spin-rate of 10 revolutions per minute.

2.2.2. Definition of performance

Performance values given in this chapter are expressed in terms of payload mass, and are based on the following main assumptions :

- launch from the CSG (French Guiana), taking account of relevant safety limitations (see para. 2.2.3).
- the launch vehicle includes : Ariane/payload adaptor type 1194 and its separation system (see para. 3.1.1.1.), and acoustic protection of the fairing (see para. 3.1.2.2.).
- the 3rd stage carries sufficient propellant to reach the intended orbit with a probability of 99.87 %.
- on jettisoning the fairing, aerothermal flux is less than 1135 W/m^2 with a probability of 99.87 % (see para. 3.4.2.4.).
- the launch vehicle does not provide for a ballistic phase.

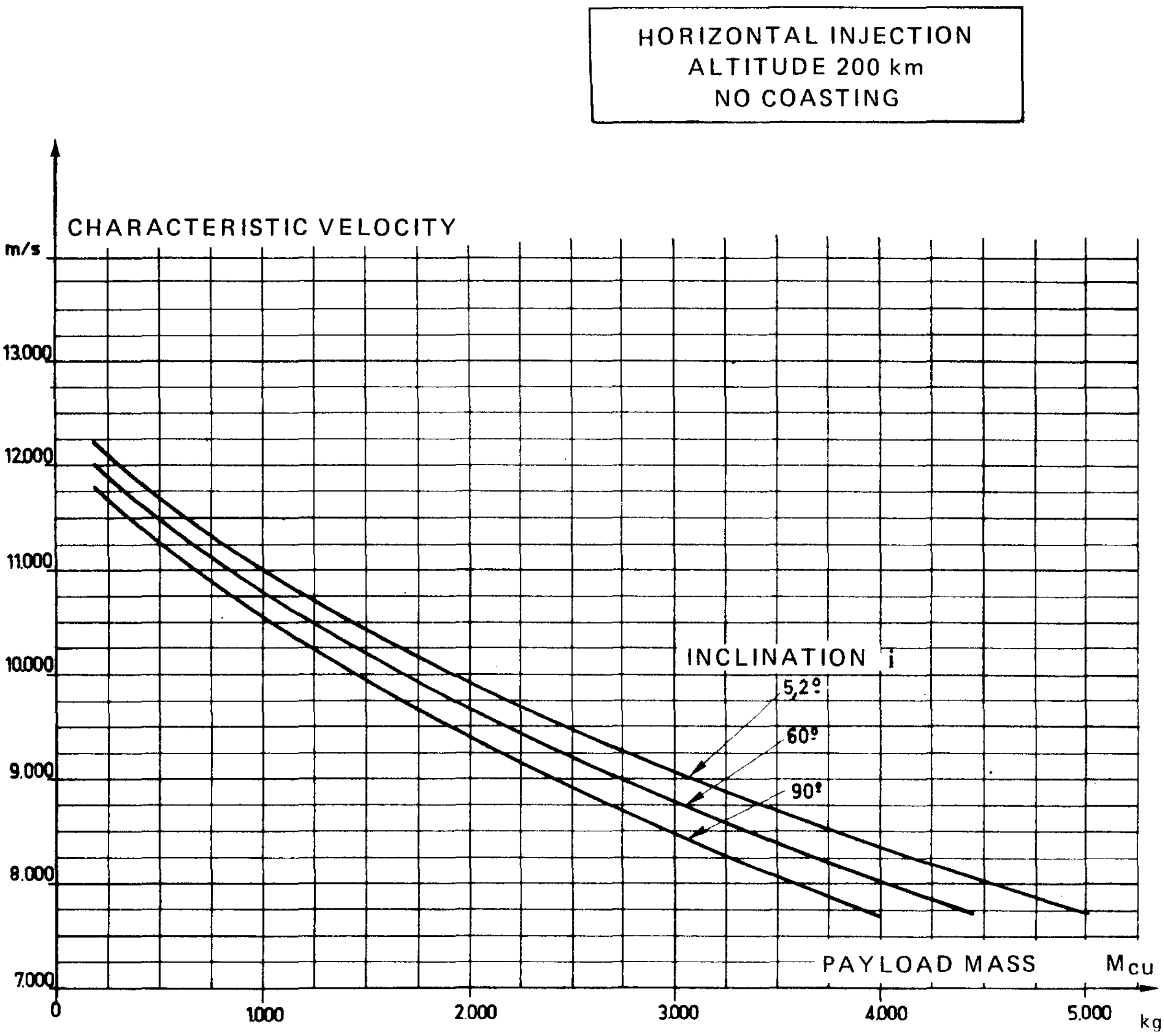


FIG. 2.1 CHARACTERISTIC VELOCITY

General performance data — Chapter 2

Performance is modified in the following cases :

- use of an adaptor other than the Ariane type 1194.
- the fairing acoustic protection not used.
- other modifications of the launch vehicle for special missions, such as :
  - addition of a fast-spin device
  - addition of a protective shield for the payload
  - possible structural reinforcement for payloads exceeding 2500 kg.

2.2.3. Launch limits

The launch azimuth for the 1st-stage flight must be between − 10.5° and + 93.5° with respect to true North. These values are imposed by the CSG safety authorities.

Certain other constraints on launch trajectories may be imposed for reasons of safety or tracking-station visibility, and must be examined case-by-case.

2.2.4. Adaptation of performance

If the payload mass is less than the Ariane launch vehicle performance for the intended orbit, the following measures may be applied :

- reduction of 3rd-stage propulsion time
- adjustment of launch vehicle trajectory
- addition of ballast
- possibility of dual or multiple launches.

2.3. General performance data

2.3.1. Characteristic velocity

Figure 2.1 shows inertial velocity available at injection, assumed to be horizontal, at an altitude of 200 km, as a function of payload mass and inclination.

The launch operation does not includes a ballistic phase, and hence this velocity is only available in the vicinity of injection.

2.3.2. Performance in circular orbit

Figure 2.2 shows payload mass as a function of altitude and inclination, for circular orbits.

2.3.3. Performance in elliptical orbit

Figure 2.3 shows elliptical-orbit performance for eastward launching, giving a final inclination of 5.2°. Figures 2.4 and 2.5 show elliptical-orbit performance, for launches giving a final inclination of 60° and 90° respectively.

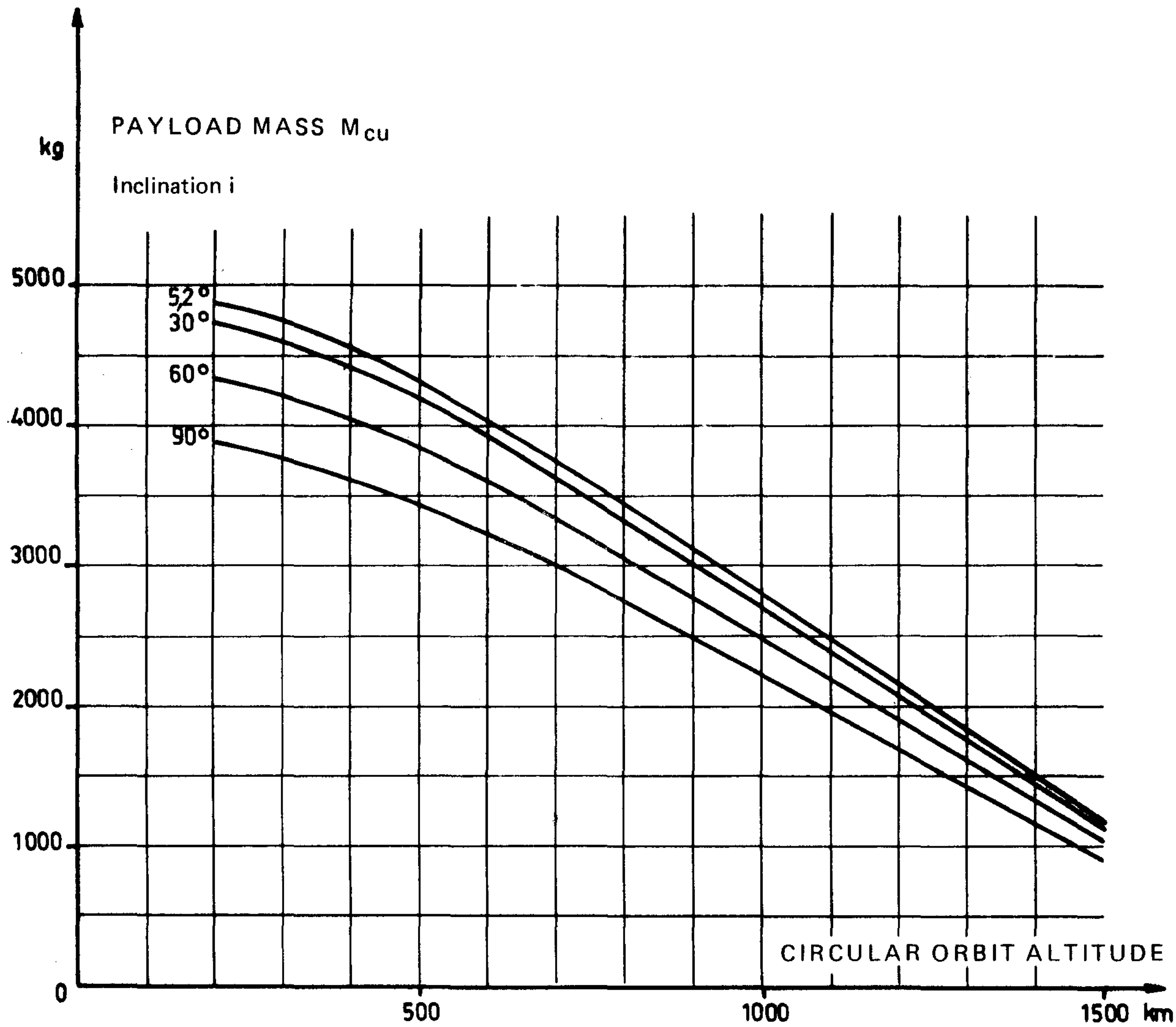


FIG. 2.2 PERFORMANCE IN CIRCULAR ORBITS

2.4. Geostationary mission

2.4.1. Performance in transfer orbit

Various performance curves are shown, in order to enable the user to determine a preliminary strategy for injection into geostationary orbit.

2.4.1.1. Variation of inclination and perigee altitude

Figure 2.6 shows the payload mass as a function of the inclination and perigee altitude of the transfer orbit, for a perigee argument of 180° (perigee on the equator) and an apogee altitude of 35 800 km.

2.4.1.2. Variation of inclination and perigee argument

Figure 2.7a shows the payload mass as a function of inclination and perigee argument, for an apogee altitude of 35 800 kg and a perigee altitude of 200 km.

For the same parameters, figures 2.7b and 2.7c show the longitude of the descending node of the transfer orbit at injection and the mean injection anomaly.

Figures 2.8a, 2.8b and 2.8c are enlargements of the framed portions of figures 2.7a, 2.7b and 2.7c respectively.

Figure 2.7a shows the limit corresponding to the 1st-stage launch azimuth constraint $\leqslant$ 93.5°.

2.4.2 Typical launch-vehicle trajectory

Figure 2.9 shows the main trajectory parameters as functions of flight time.

2.4.3. Influence of miscellaneous parameters

The values indicated below are valid for the orbit range shown in figure 2.8a.

2.4.3.1. Perigee and apogee

The derivatives are as follows :

$$\frac{\partial Mcu}{\partial Z_a} = -\ 0.016 \text{ kg/km} \ ; \qquad \frac{\partial Mcu}{\partial Z_p} = -\ 0.66 \text{ kg/km}$$

where
Z_a = apogee altitude
Z_p = perigee altitude

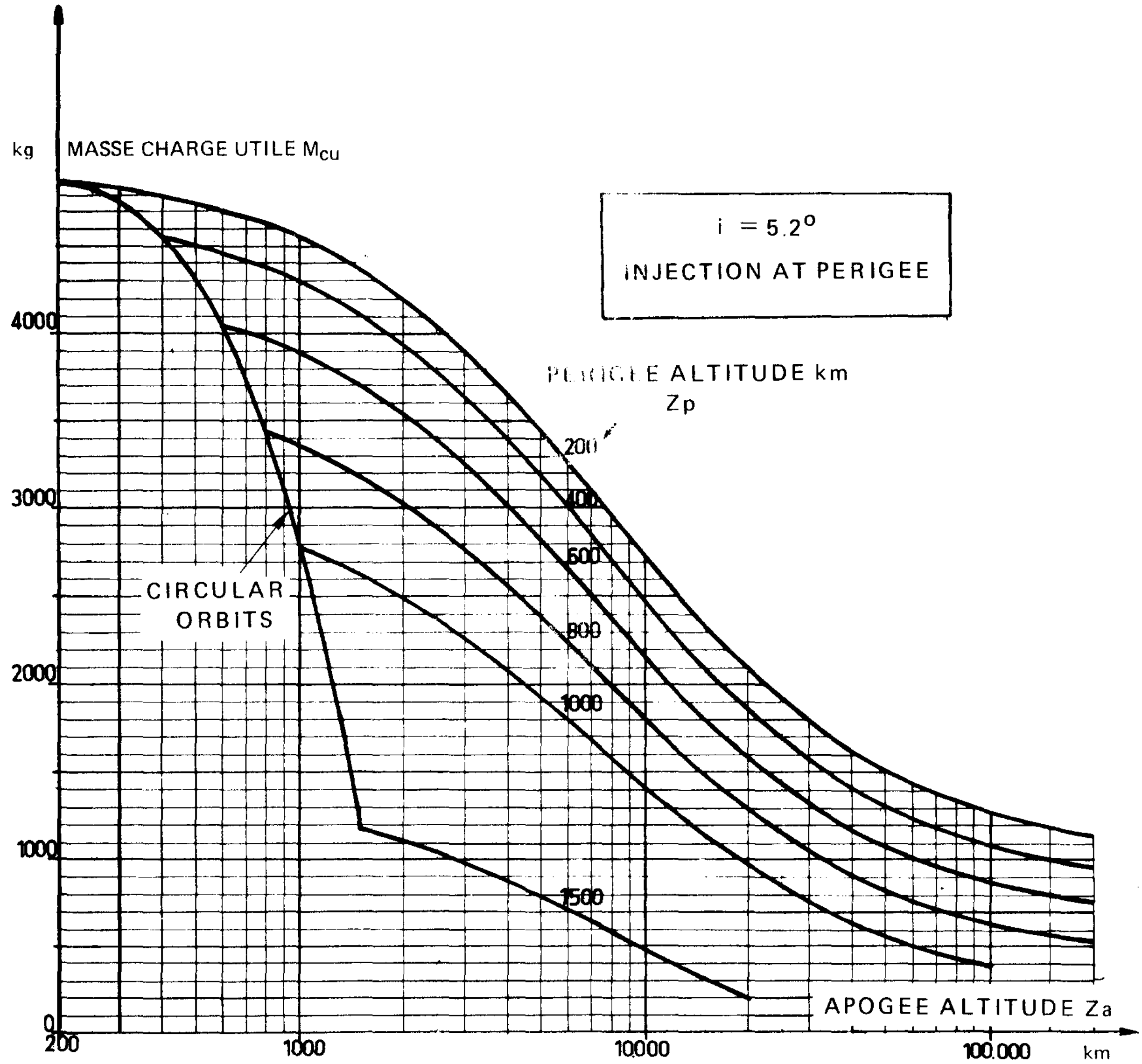


Fig. 2.3 PERFORMANCE IN ELLIPTICAL ORBITS
LAUNCH AZIMUTH 90°
INCLINATION 5.2°

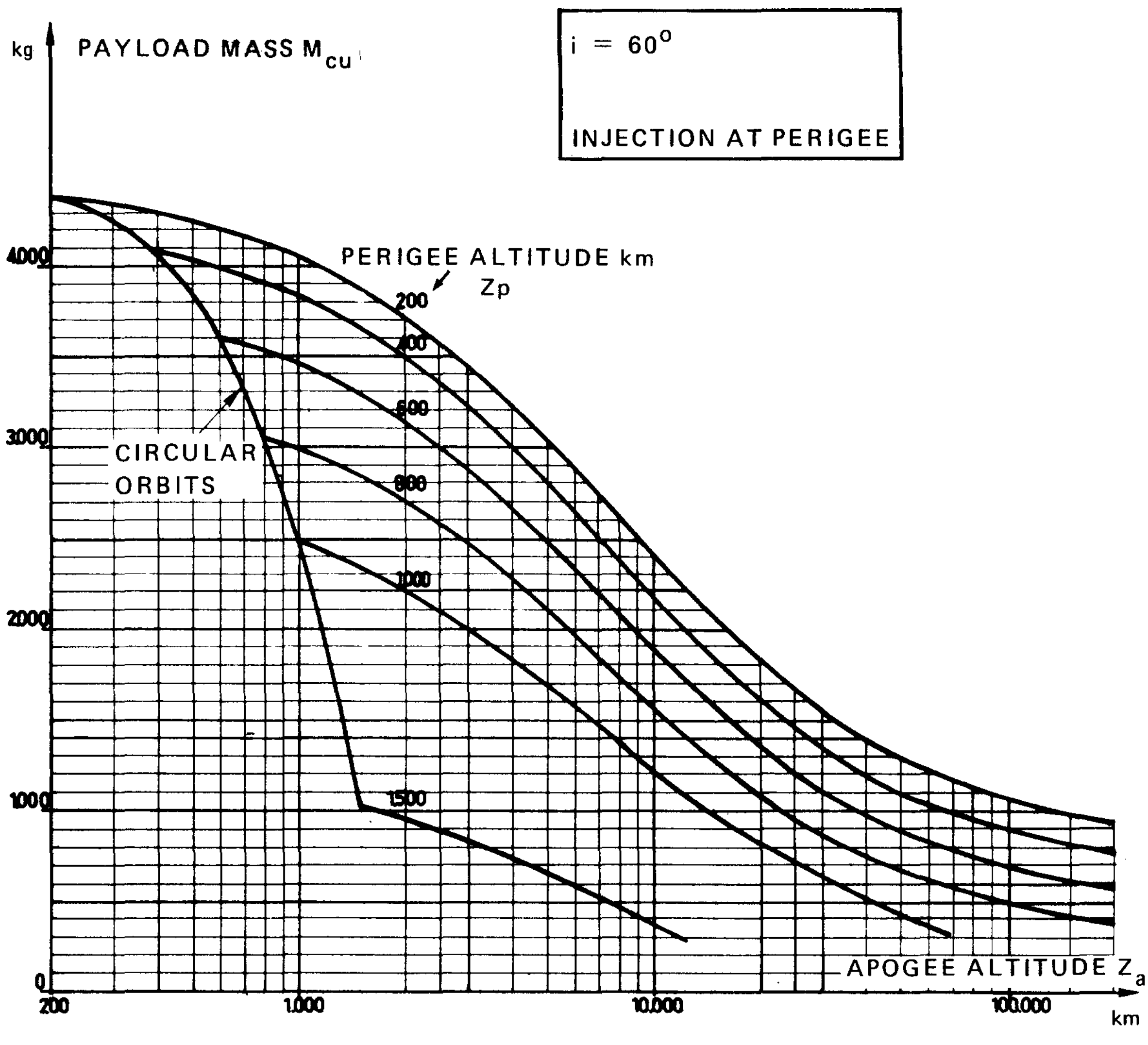


FIG. 2.4 PERFORMANCE IN ELLIPTICAL ORBITS INCLINATION 60°

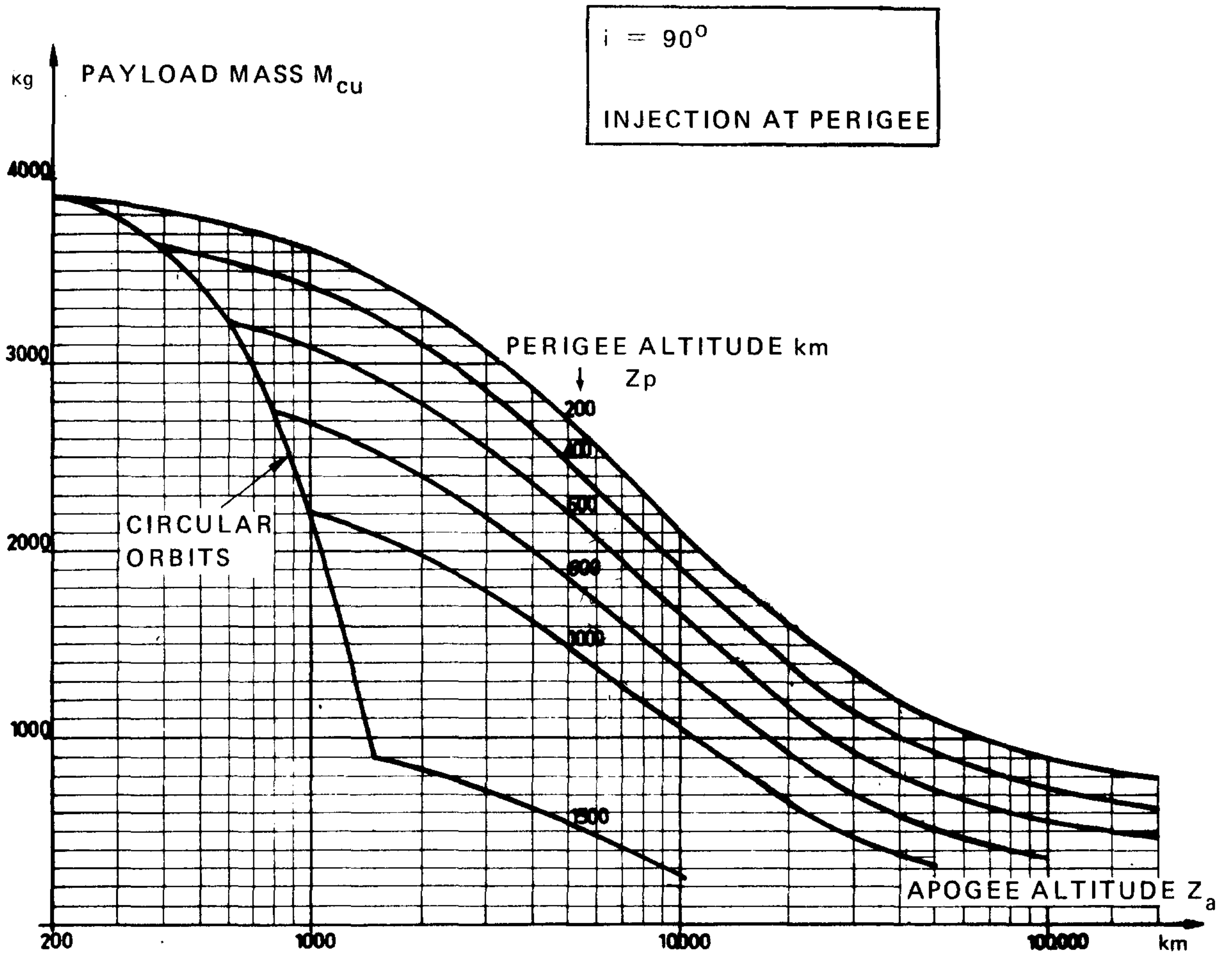


FIG. 2.5 PERFORMANCE IN ELLIPTICAL ORBITS INCLINATION 90°

Za : 35 800 km
180°
LAUNCH AZIMUTH 93.5°

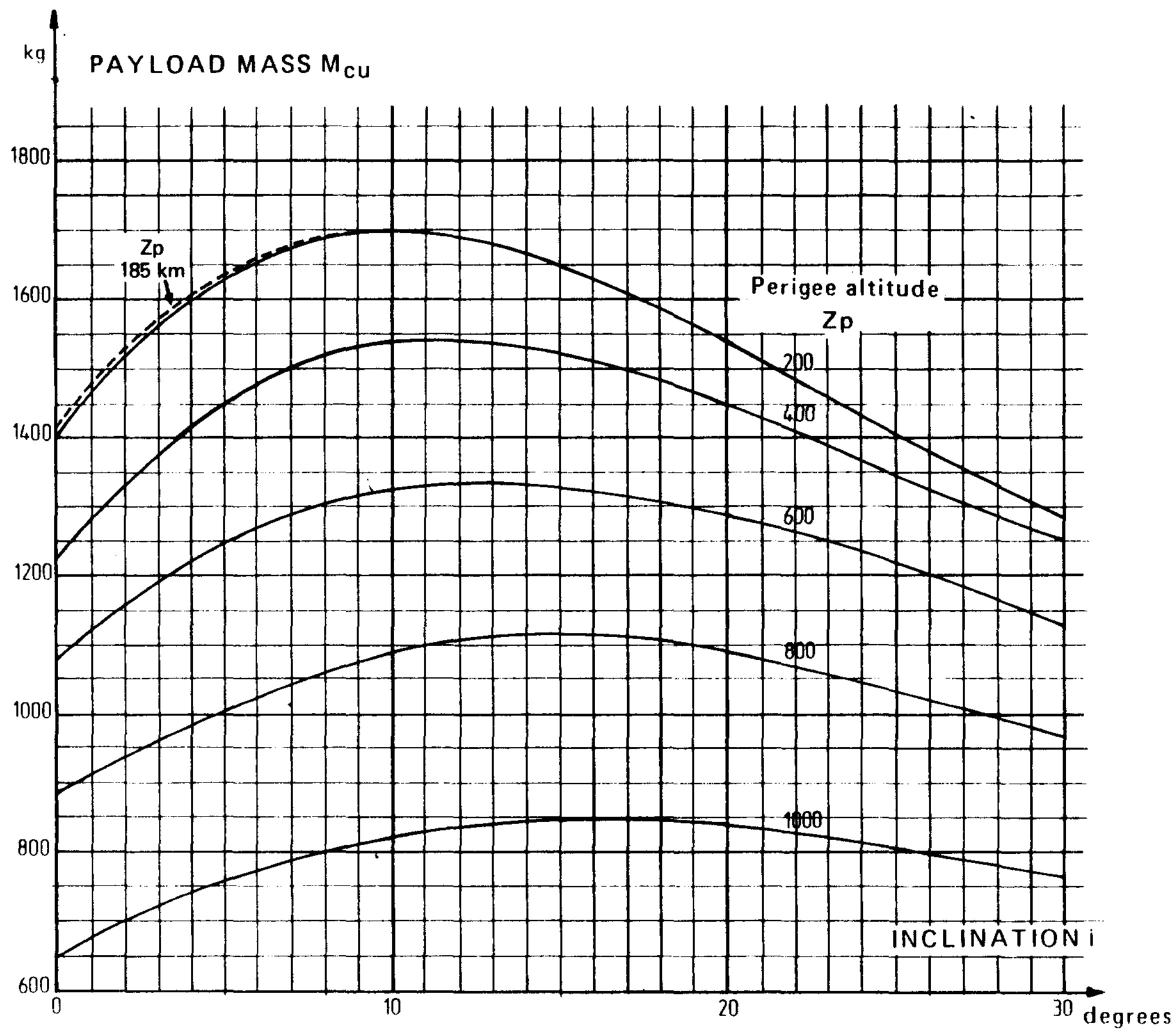


FIG. 2.6 PERFORMANCE IN GEOSTATIONARY TRANSFER ORBIT
$M_{cu} = f(i, Z_p)$

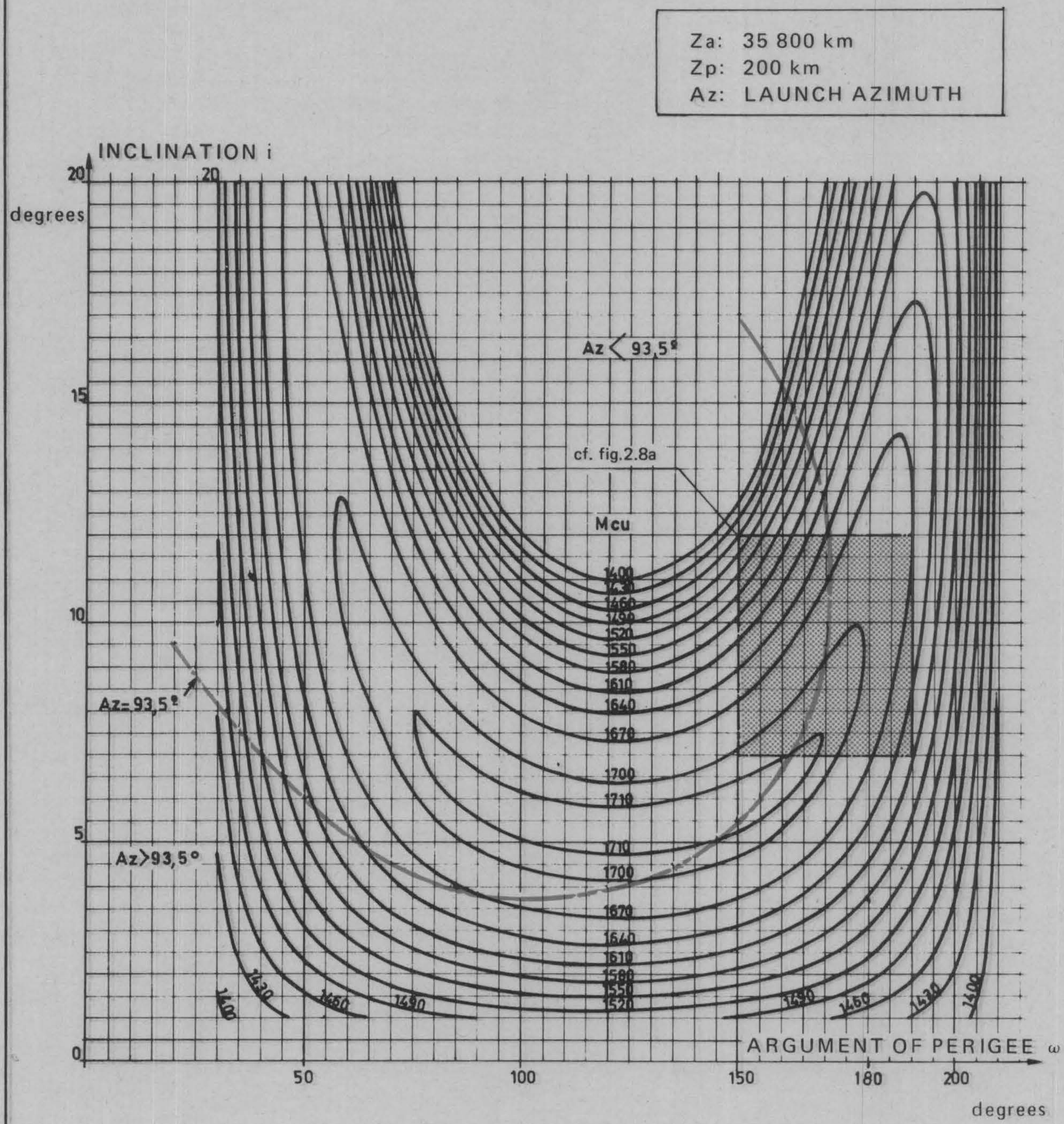


FIG. 2.7 a PERFORMANCE IN GEOSTATIONARY TRANSFER ORBIT
$M_{cu} = f(i, \omega)$

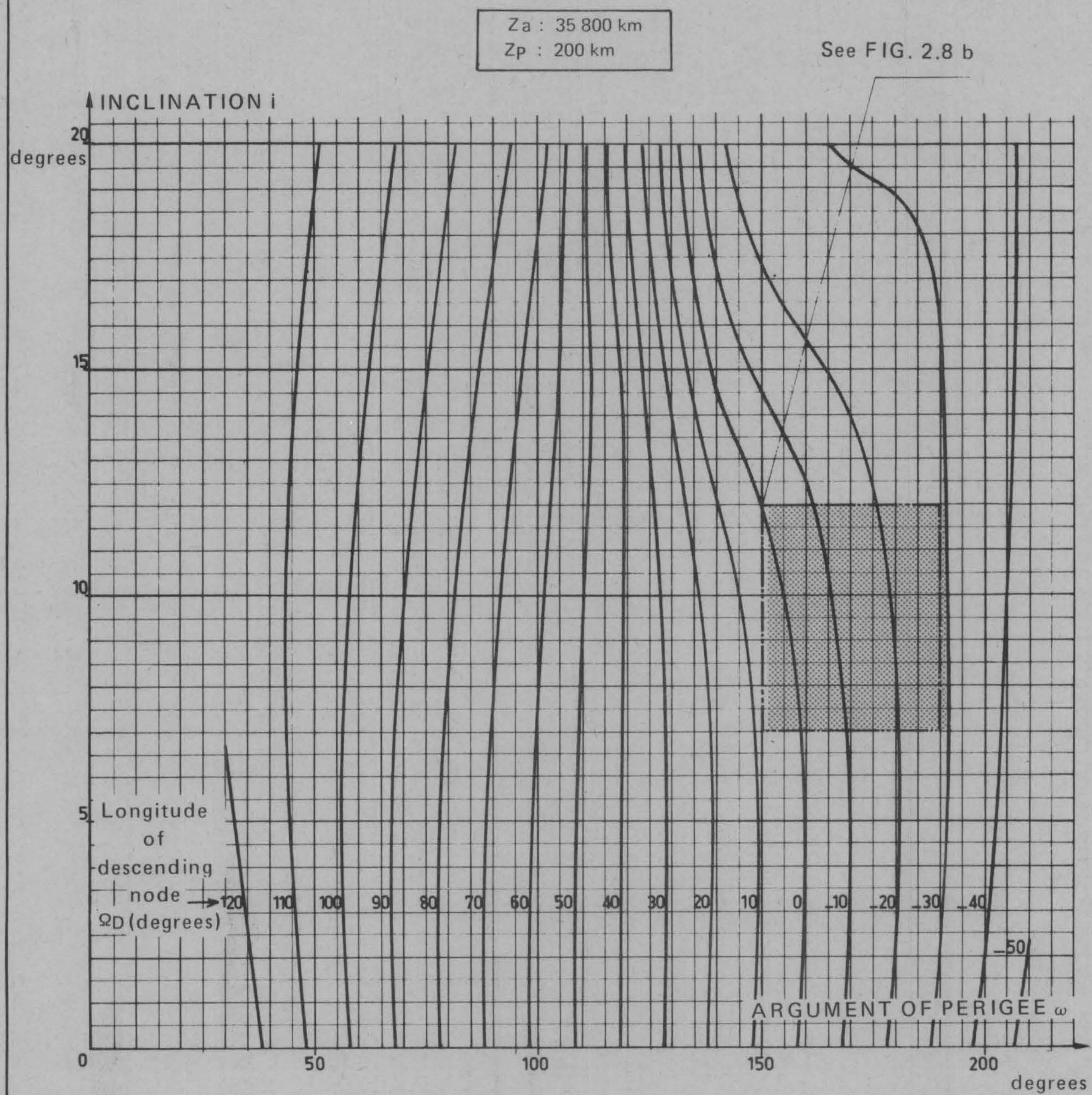


FIG. 2.7 b PERFORMANCE IN GEOSTATIONARY TRANSFER ORBIT

$\Omega_D = f(i, \omega)$

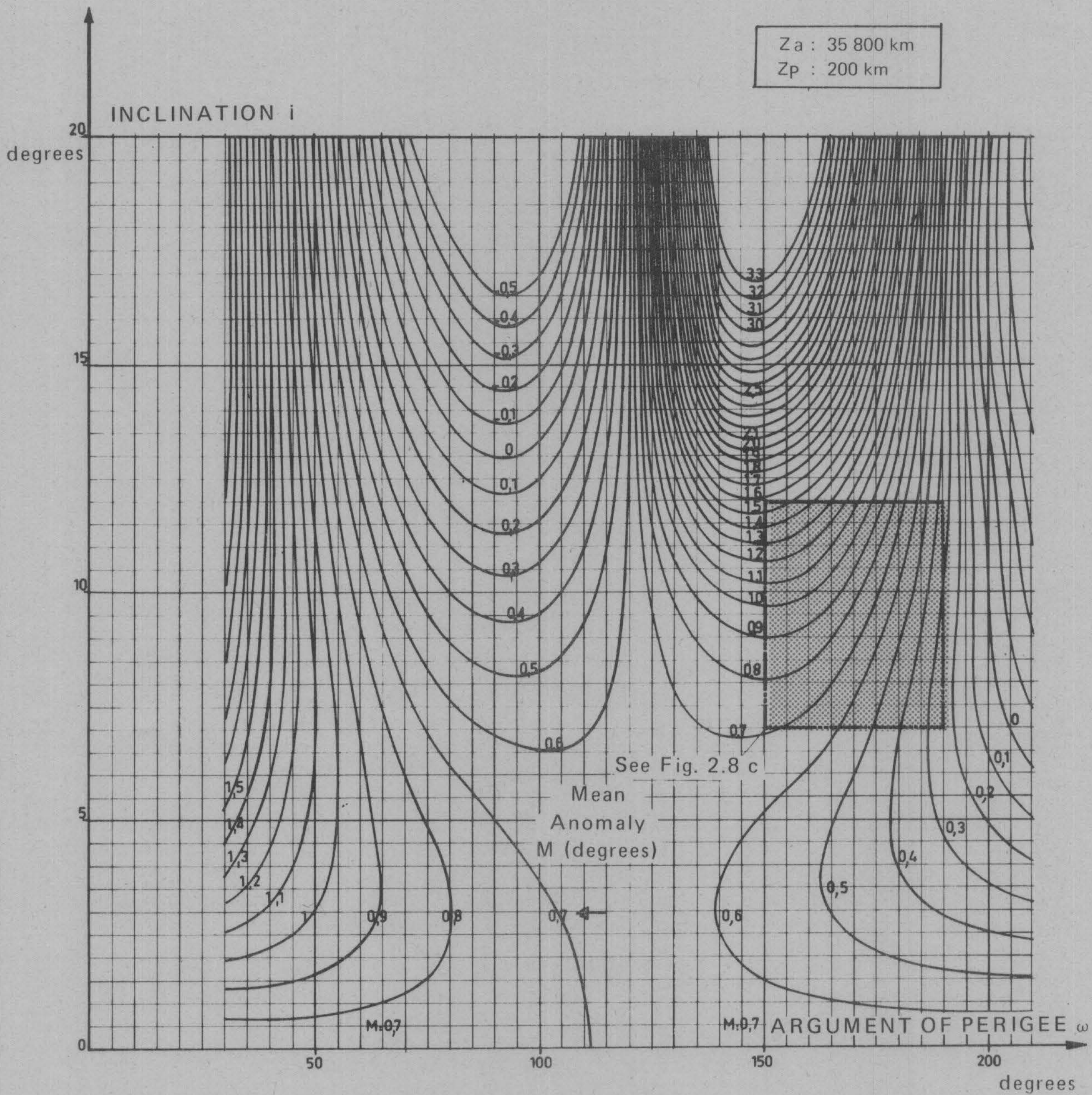


FIG. 2.7 c PERFORMANCE IN GEOSTATIONARY TRANSFER ORBIT
$M = f(i, \omega)$

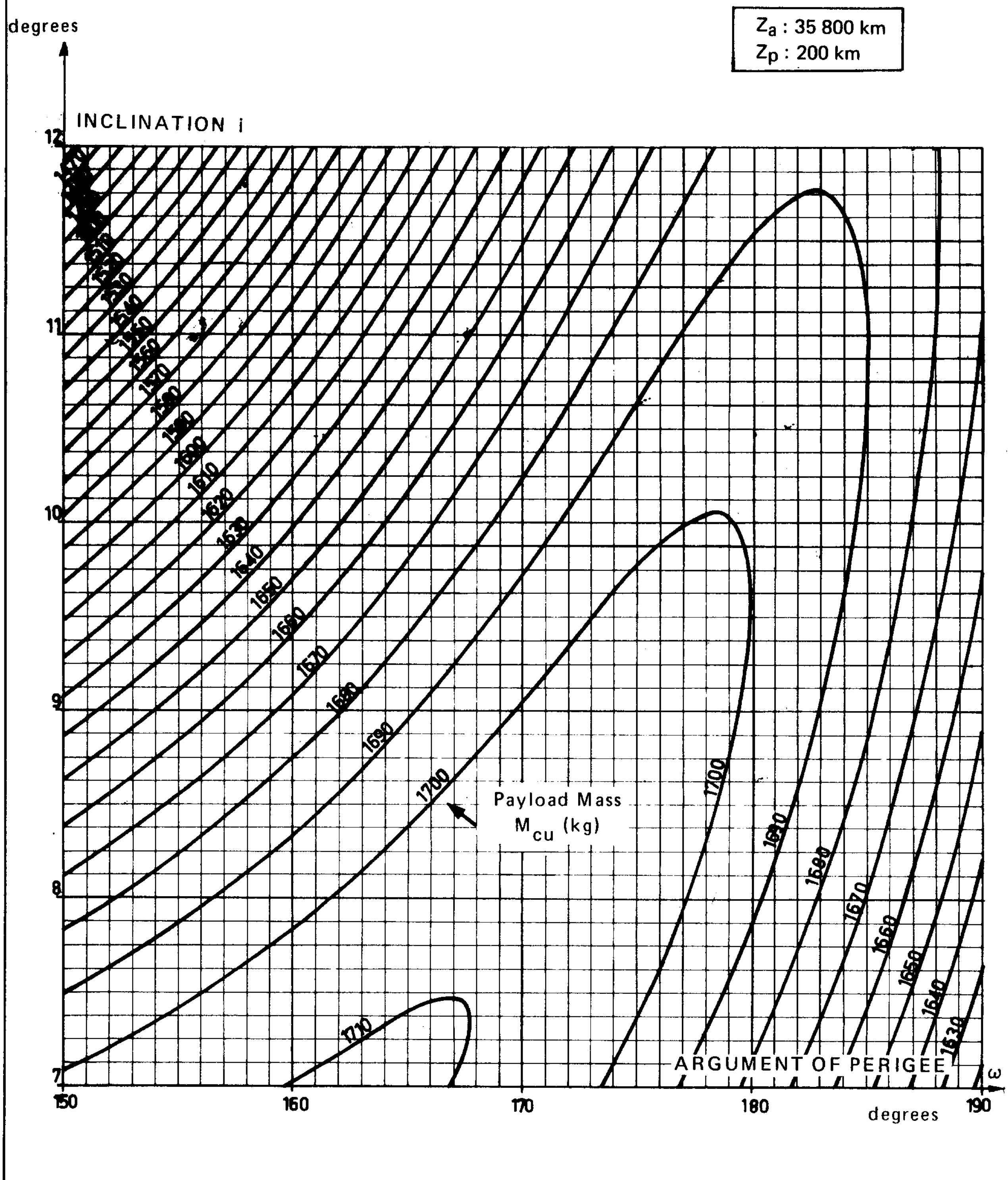


FIG. 2.8 a PERFORMANCE IN GEOSTATIONARY TRANSFER ORBIT
Payload Mass $M_{cu} = f(i, \omega)$

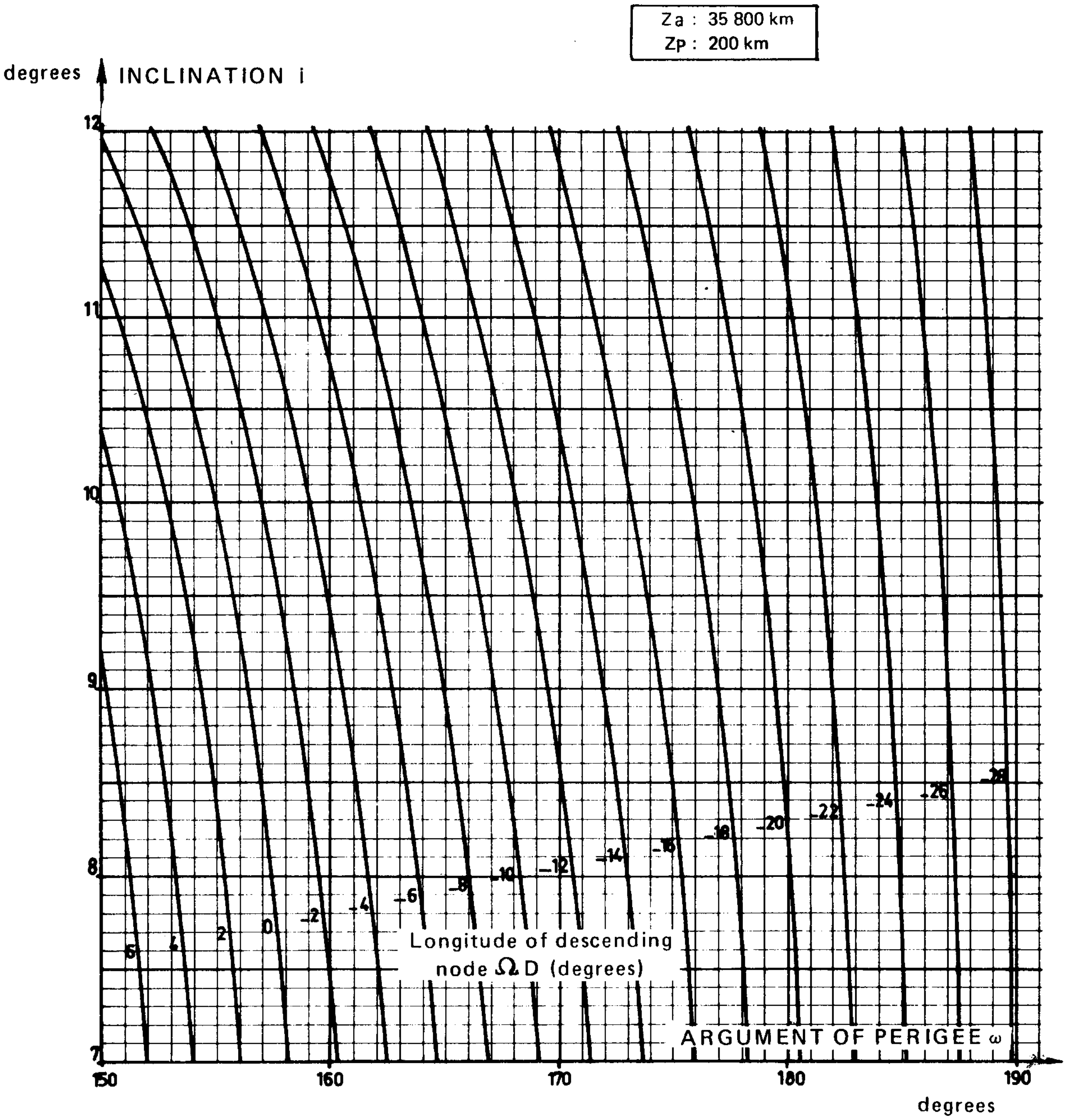


FIG. 2.8 b PERFORMANCE IN GEOSTATIONARY TRANSFER ORBIT

$\Omega_D = f(i, \omega)$

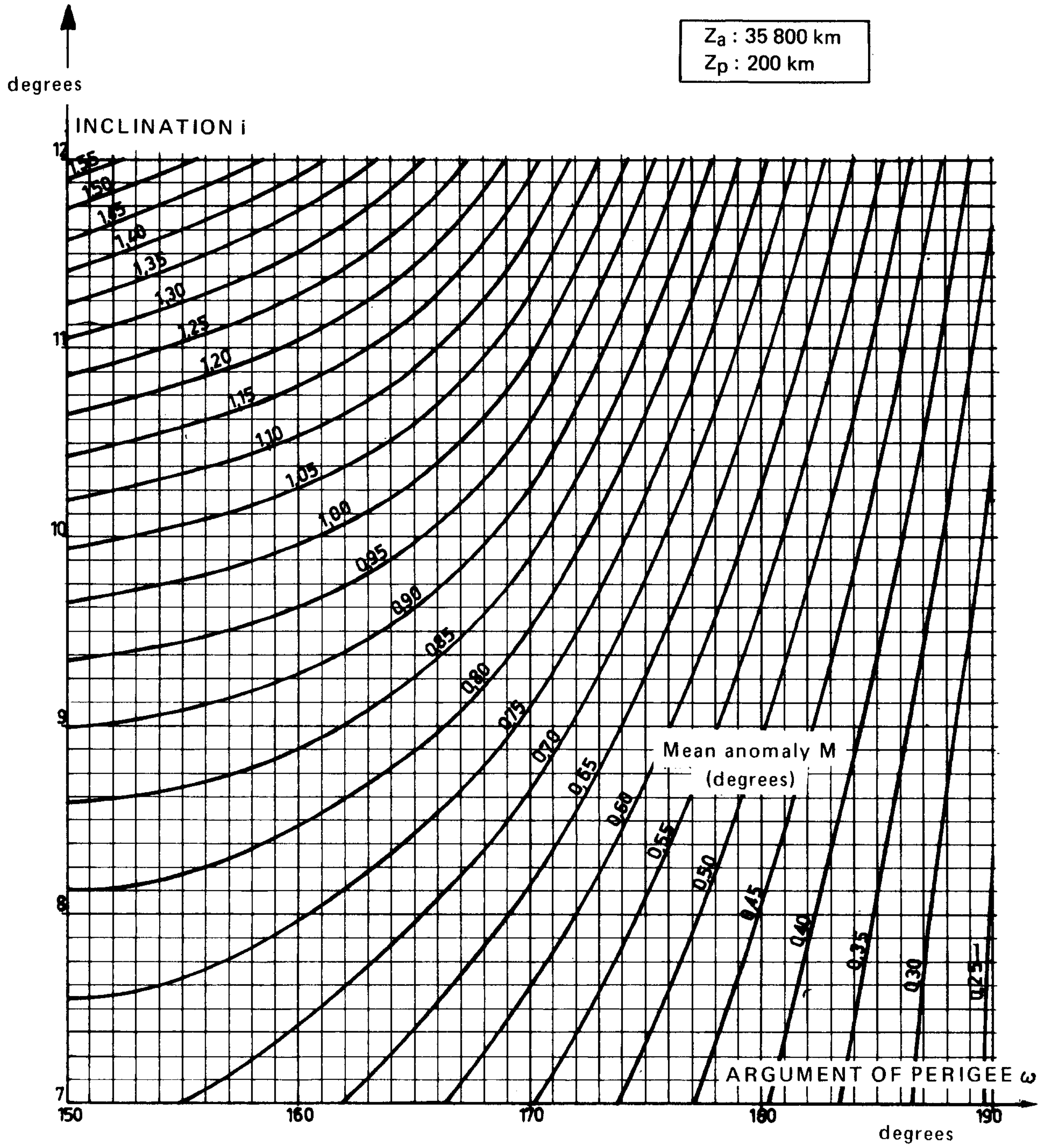


FIG. 2.8 c PERFORMANCE IN GEOSTATIONARY TRANSFER ORBIT
Mean anomaly M = f(i, ω)

ORBIT

a	=	24 371 km	Z	=	ALTITUDE
e	=	0.73	X	=	GROUND RANGE
i	=	9.65^{o}	V_R	=	RELATIVE VELOCITY
ω	=	180^{o}	Γ	=	LONGITUDINAL ACCELERATION
Ω	=	-144.6^{o}	θ_L	=	LOCAL PITCH ANGLE
Z_a	=	35 786 km			
Z_p	=	200 km			

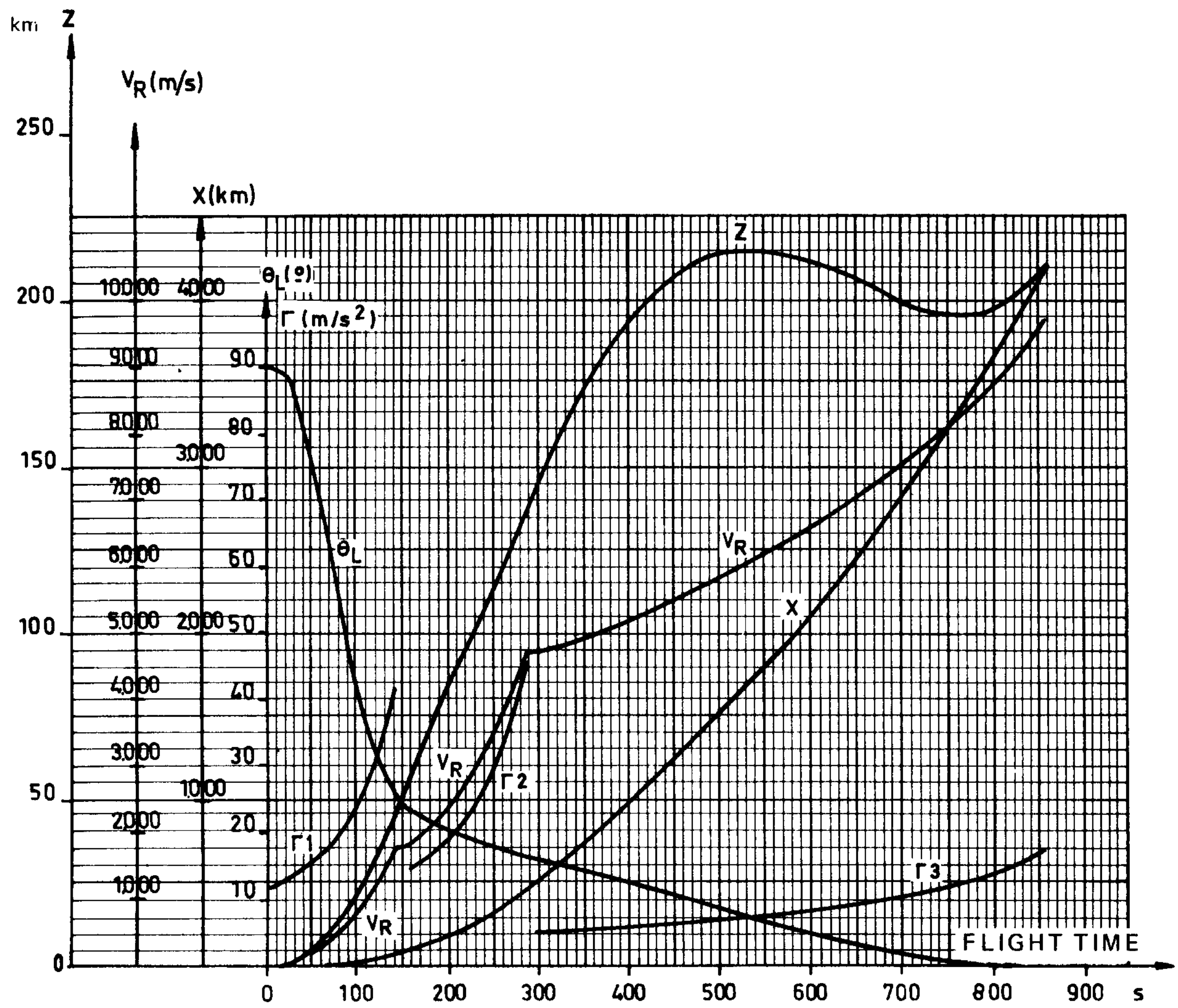


FIG. 2.9 TYPICAL TRAJECTORY – GEOSTATIONARY MISSION

2.4.3.2. Aerothermal flux at fairing jettison

Aerothermal flux is nominally 1135 W/m², with a probability of 99.87 %, at fairing jettison (see para. 3.4.2.4.).

However, the instant of jettisoning can be readjusted for special missions, in which case the flux value is different. For example, for an aerothermal flux of 2270 W/m², there is a payload gain of 10 kg.

2.4.3.3. Descending node

The performance values shown in figure 2.8 a are calculated without constraint on the orbit node. The values of this parameter are as indicated in figure 2.8 b. The launch vehicle can make adjustments with respect to these values, in which case the resultant performance loss can be calculated approximately by the following equation :

$$\Delta Mcu_{kg} = -|\Delta \Omega|^2 \quad (\Omega \text{ in degrees})$$

2.4.3.4. Acoustic protection

If acoustic protection is omitted, a payload mass gain of 7 kg is achieved.

2.4.4. Injection accuracy

The covariance matrix of injection errors on the orbital parameters is as follows (for orbits defined in fig. 2.8 a) :

a	e	i	ω	Ω	
463	0.00519	− 0.0151	− 0.0126	0.0557	a
	$5.85\ 10^{-8}$	$-1.59\ 10^{-7}$	$2.0\ 10^{-7}$	$5.75\ 10^{-7}$	e
		0.000351	0.00140	− 0.00141	i
			0.00595	− 0.00572	ω
				0.00576	Ω

This gives the following standard deviation values :

a	Semi-major axis	21.5 km
e	Eccentricity	0.00024
i	Inclination	0.019°
ω	Perigee argument	0.077°
Ω	Ascending node	0.076°
Z_p	Perigee altitude	0.45 km
Z_a	Apogee altitude	43.1 km

Parameter M_0 (mean injection anomaly) depends on the flight sequence, and its accuracy is not correlated with the other parameters. Standard deviation : TBD.

2.5. Elliptical-orbit mission

2.5.1. Performance

Elliptical-orbit performance details are given in paragraph 2.3.3. Figure 2.10 shows elliptical-orbit performance for a perigee of 300 km and an apogee of 100 000 km, as a function of inclination and perigee argument.

2.5.2. Typical launch-vehicle trajectory

Figure 2.11 gives the main parameters for a trajectory having a perigee of 300 km and an apogee of 11 241 km as a function of flight time.

2.5.3. Elliptical-orbit accuracy

Figure 2.12 shows the dispersions of the orbital parameters for the elliptical orbits in figures 2.3 to 2.5.

2.6. Sun-synchronous mission

2.6.1. Performance

For a sun-synchronous orbit, inclination depends on the geometrical characteristics of the orbit. Performance details shown in figure 2.13 take account of this requirement, showing both payload mass and inclination for a variable circular-orbit altitude. Figure 2.14 shows ascending-node longitude, and injection longitude and latitude as a function of circular-orbit altitude.

2.6.2. Typical launch-vehicle trajectory

Figure 2.15 gives the main trajectory parameters as functions of flight time.

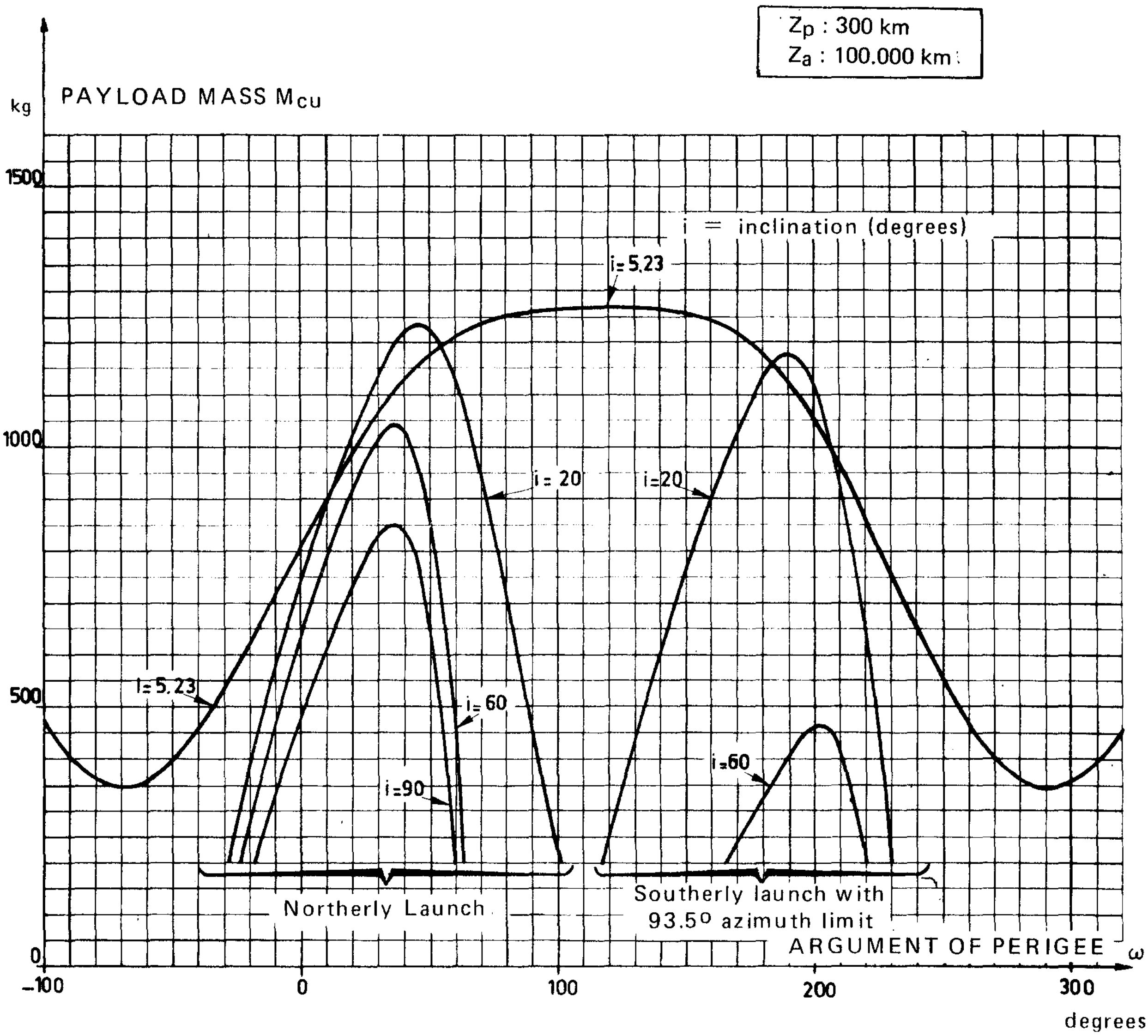


FIG 2.10 PERFORMANCE IN ELLIPTICAL ORBITS

$M_{cu} = f(i, \omega)$

ORBIT

a	=	12 149 km	Z	=	ALTITUDE
e	=	0.45	X	=	GROUND RANGE
i	=	5.23^{o}	V_R	=	RELATIVE VELOCITY
ω	=	124.28^{o}	Γ	=	LONGITUDINAL ACCELERATION
Ω	=	-91.33^{o}			
Z_a	=	11 241 km			
Z_p	=	300 km	θ_L	=	LOCAL PITCH ANGLE

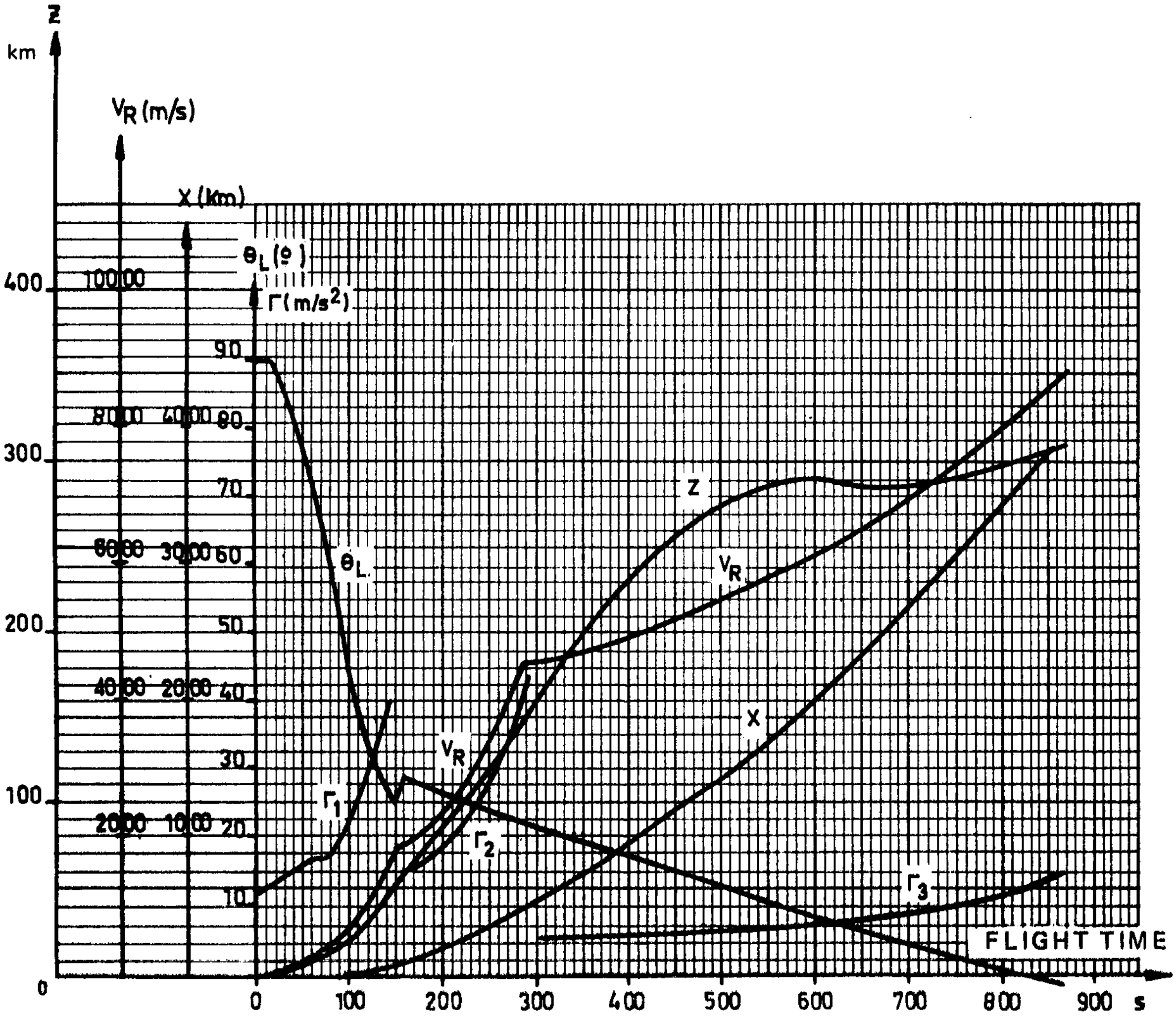


FIG. 2.11 TYPICAL TRAJECTORY
ELLIPTICAL ORBIT MISSION

FIG. 2.12 DISPERSIONS FOR ELLIPTICAL ORBITS
(Will be published in a subsequent issue)

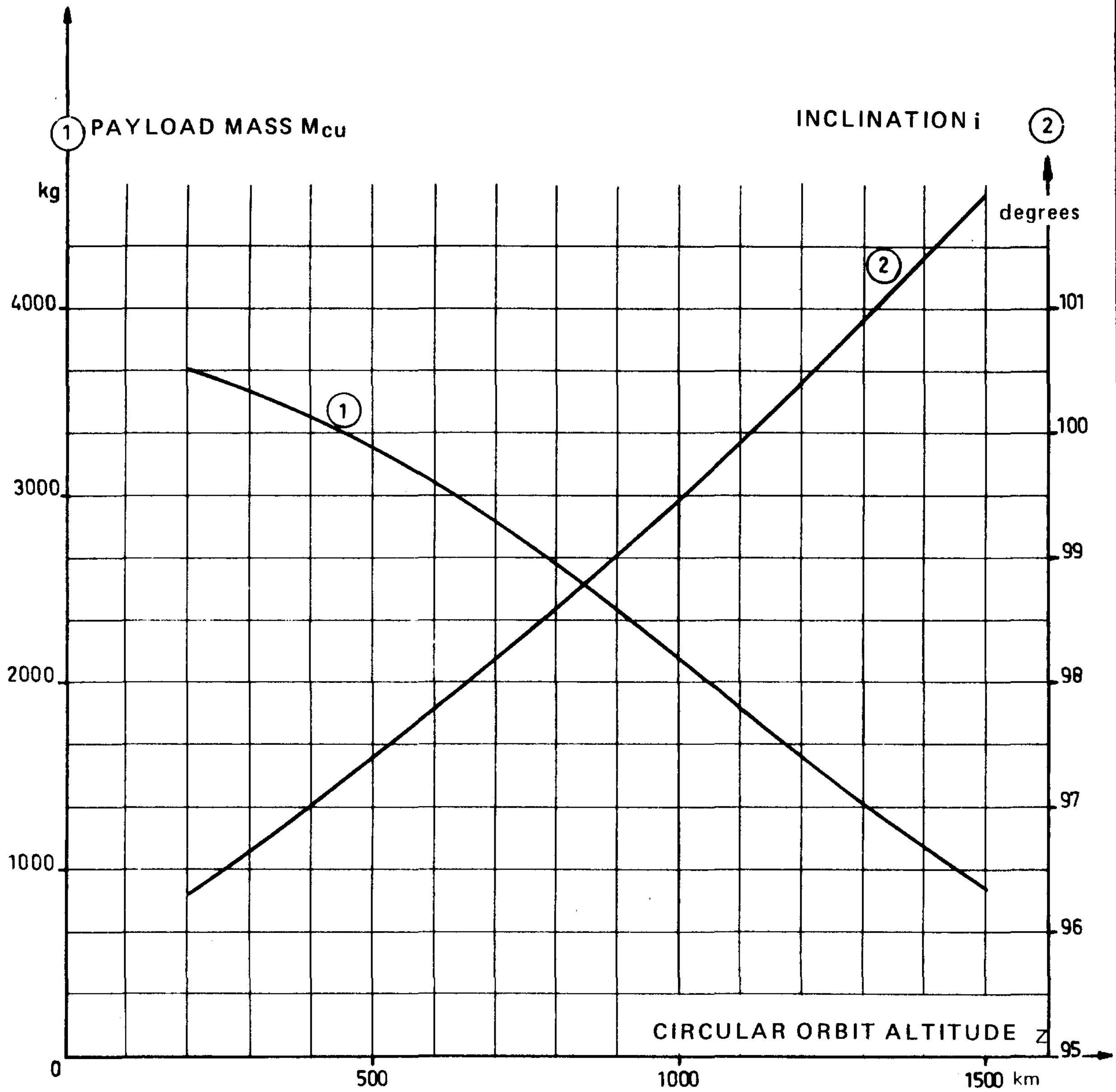


FIG. 2.13 PERFORMANCE IN SUN–SYNCHRONOUS ORBITS

$M_{cu} = f(Z)$

$i = f(Z)$

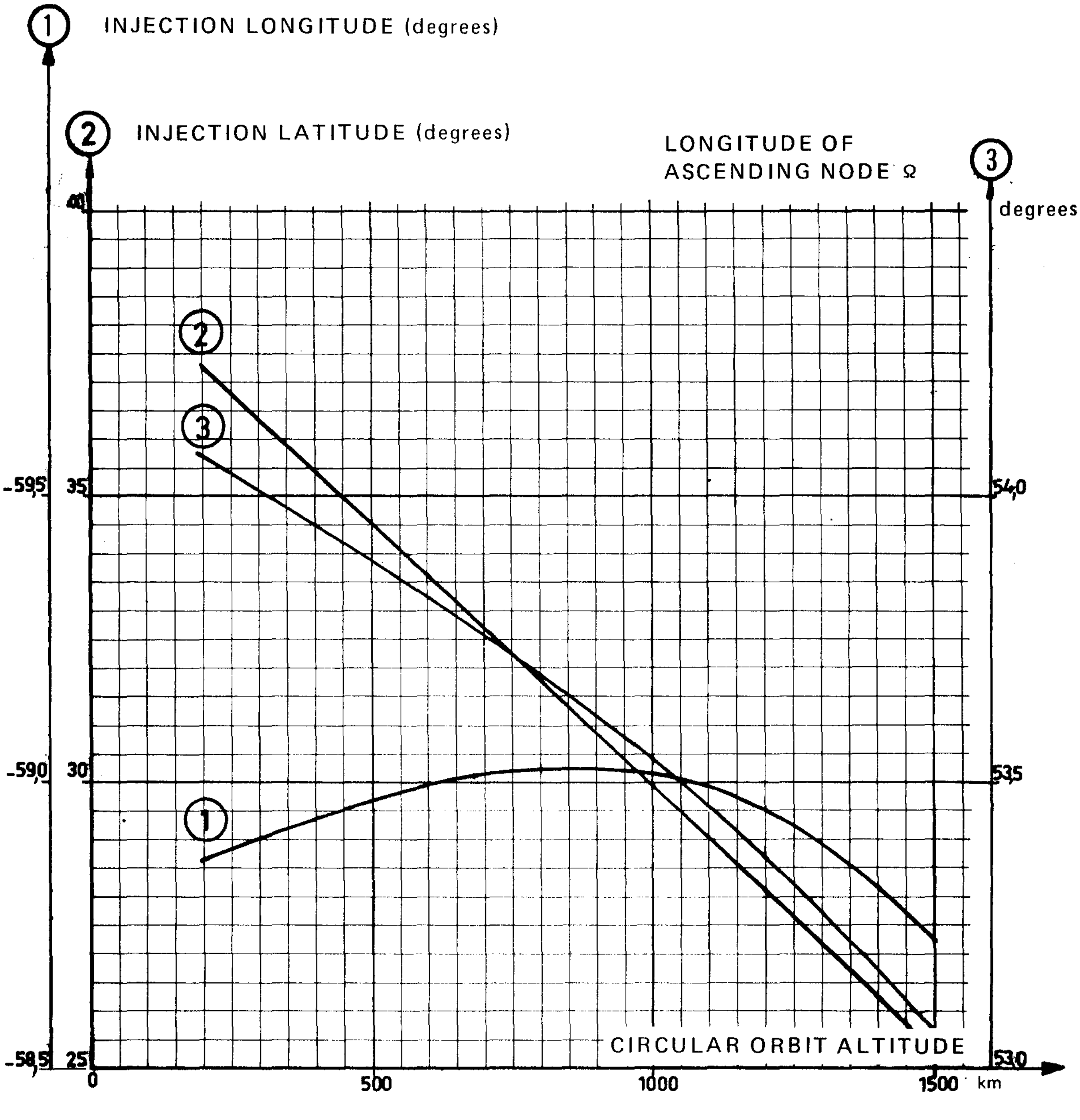


FIG. 2.14 PERFORMANCE IN SUN–SYNCHRONOUS ORBITS
LATITUDE AND LONGITUDE AT INJECTION
LONGITUDE OF ASCENDING NODE

ORBIT

a	=	7 218 km	Z	=	ALTITUDE
e	=	0	X	=	GROUND RANGE
i	=	98.76°	V_R	=	RELATIVE VELOCITY
Ω	=	2.69°	Γ	=	LONGITUDINAL ACCELERATION
			θ_L	=	LOCAL PITCH ANGLE

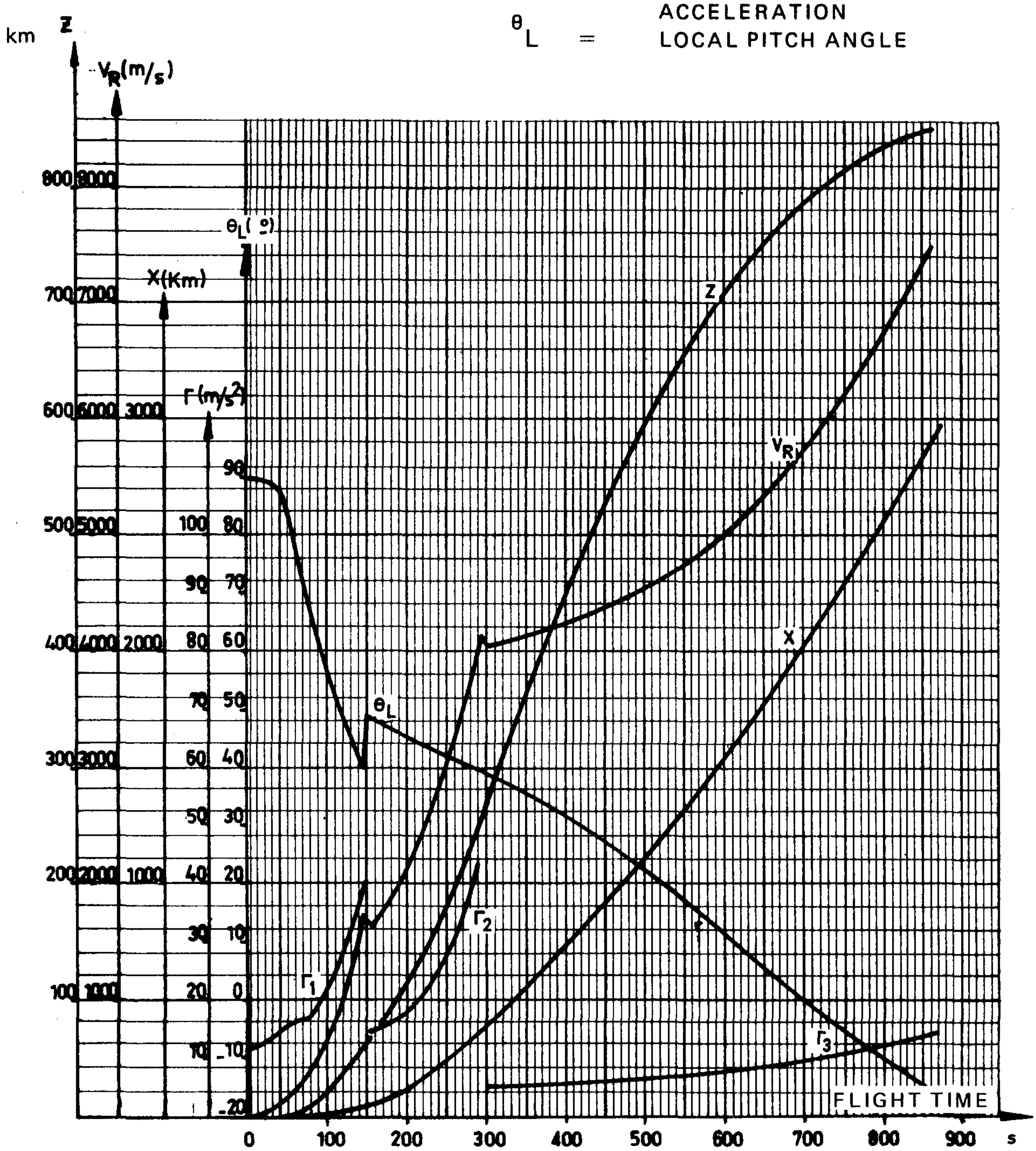


FIG. 2.15 TYPICAL TRAJECTORY
SUN–SYNCHRONOUS MISSION

2.6.3. Injection accuracy

For the following typical syn-synchronous orbit :

a = 7208.9 km
e = 0.002
i = 98.73°
Ω = 2.77°

The covariance matrix of injection errors is as follows :

a	e	i	Ω	
1.40	0.000141	− 0.00718	0.007 10	a
	$2.83\ 10^{-7}$	$-4.97\ 10^{-6}$	$-6.21\ 10^{-8}$	e
		0.000698	$6.92\ 10^{-5}$	i
			$7.96\ 10^{-6}$	Ω

This gives the following standard deviations for the orbital parameters :

a	Semi-major axis	1.18 km
e	Eccentricity	0.00017
i	Inclination	0.0264°
Ω	Ascending node	0.0028°
Z_a	Apogee altitude	2.22 km
Z_p	Perigee altitude	0.91 km

2.7. Escape mission

2.7.1. Performance

Performance data for escape missions are given in figure 2.16 as functions of parameter C_3 (square of velocity at infinity) and the declination of velocity at infinity.

These performance data are given for a hyperbola perigee altitude of 300 km, and for optimum launch time.

2.7.2. Typical launch-vehicle trajectory

Figure 2.17 gives the main trajectory parameters as functions of flight time.

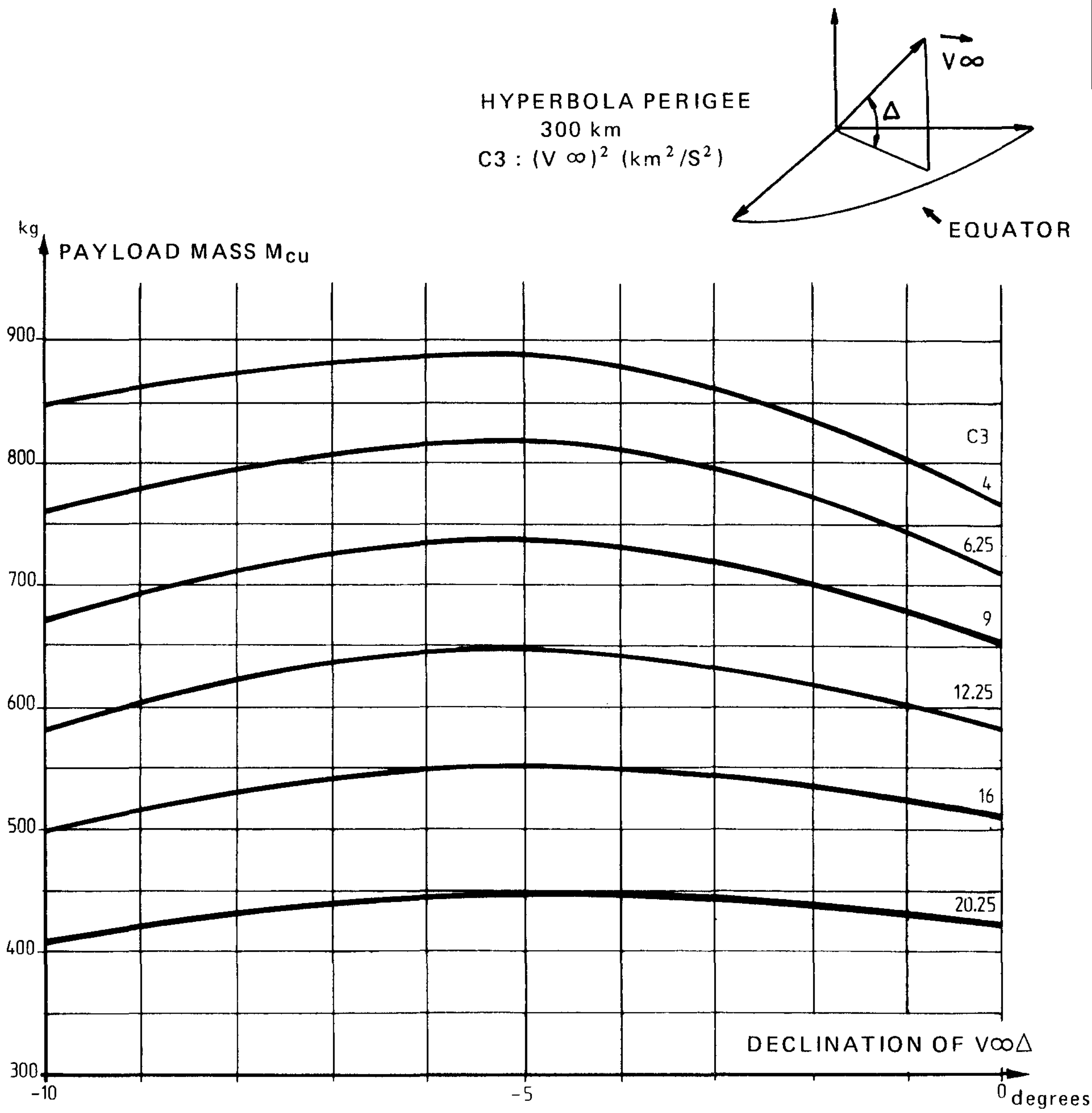


FIG. 2.16 PERFORMANCE IN ESCAPE TRAJECTORIES

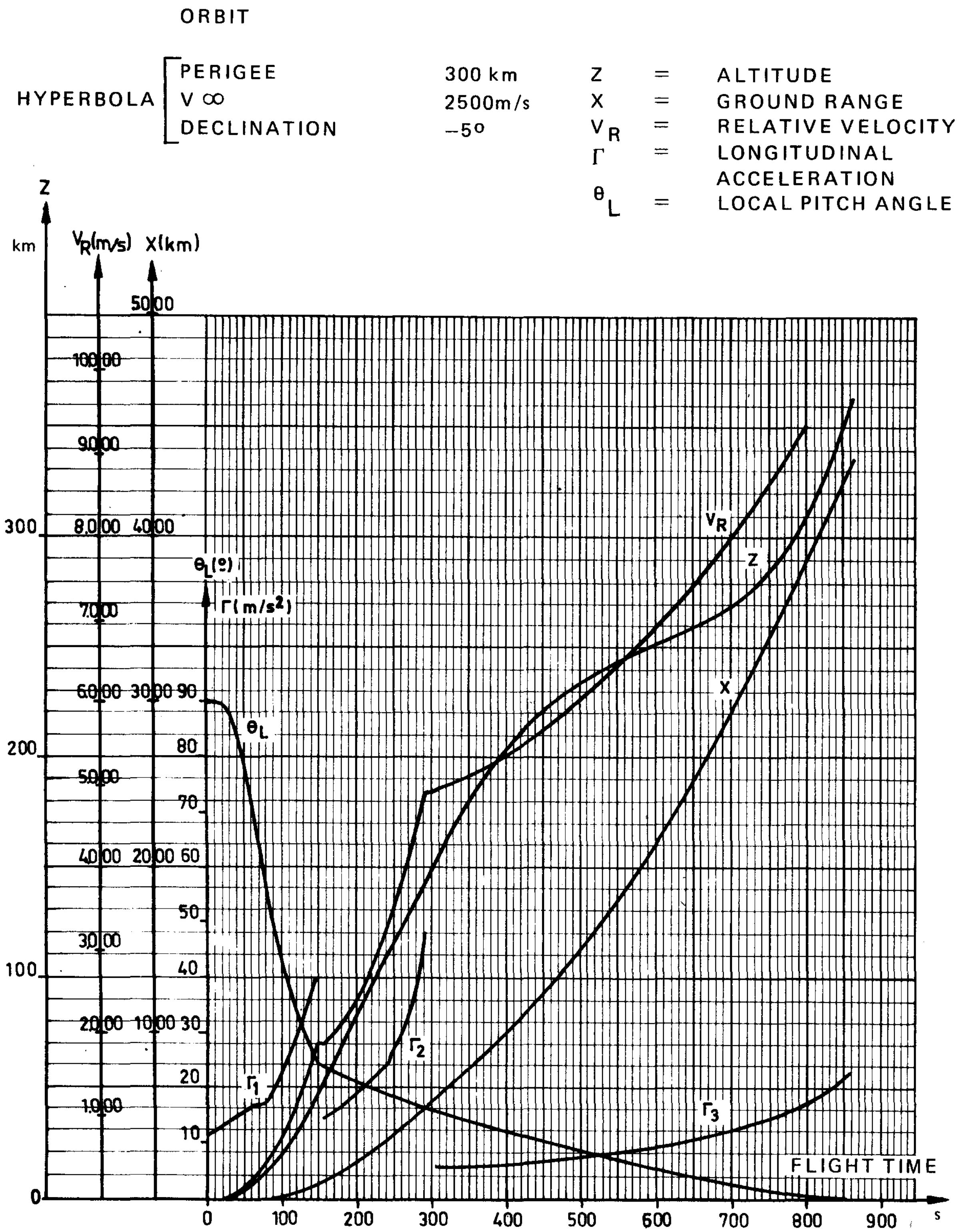


FIG. 2.17 TYPICAL TRAJECTORY IN ESCAPE MISSION

2.7.3. Variation of launch time

For escape missions, the direction of hyperbola velocity at infinity is fixed (declination and right ascension). For a non-optimum launch time, the hyperbola plane varies, producing a change in the launch-vehicle ascent trajectory.

The Ariane inertial-guidance system is designed to permit this variation, and allow a daily launch window. This causes a slight resultant decrease of performance, as indicated in the following table (in relation to optimum time) :

Δt	− 20 mn	− 10 mn	+ 10 mn	+ 20 mn
ΔMcu	− 26 kg	− 6.5 kg	− 4.8 kg	− 20 kg

The values indicated are valid for the trajectory described in para. 2.7.2.

For trajectories having the same perigee, and similar payload mass, these values remain largely valid.

2.7.4. Injection accuracy

For the trajectory shown in figure 2.17, the covariance matrix of injection errors for the following parameters is given in the table below :

V : absolute velocity at injection (m/s)
Z : altitude at injection (km)
θ : angle of absolute velocity with the horizontal (degrees)
Az : angle of horizontal velocity with North (degrees)

V	Z	θ	Az	
1,04	− 0.279	− 0.00222	0.000130	V
	0.278	0.00272	$-7.91\ 10^{-5}$	Z
		$3.1\ 10^{-5}$	$-4.53\ 10^{-7}$	θ
			0.000297	Az

Nominal values of these parameters and their standard deviations are as indicated below :

	Nominal value	Standard deviation
V	11160.6 m/s	1.02 m/s
Z	364.9 km	0.527 km
θ	5,69°	0.0056°
Az	93.39°	0.0172°

2.8. Description of typical geostationary-mission flight

2.8.1. Flight profile

See figure 2.18

2.8.2. Flight sequence (seconds)

0 :	ignition of 1st-stage engines (H0)
3.4 :	liftoff
23.0 :	end of vertical ascent, tiltover
85.0 :	maximum thrust
145.8 :	detection of 1st-stage burnout (H1)
150.6 :	1st stage separates
150.9 :	ignition of 2nd-stage engine
153.8 :	2nd-stage engine attains maximum thrust
248.0 :	fairing jettison
288.2 :	detection of 2nd-stage burnout (H2)
293.1 :	2nd stage separates
295.1 :	ignition of 3rd-stage engine
298.2 :	3rd-stage engine reaches maximum thrust
845.0 :	3rd-stage thrust cutoff commanded (H3)
847.0 :	start of attitude control (payload orientation)
903.0 :	end of orientation and start of spinup (10 rpm)*
949.0 :	payload separates (H4)

* spinup optional (see para 3.3).

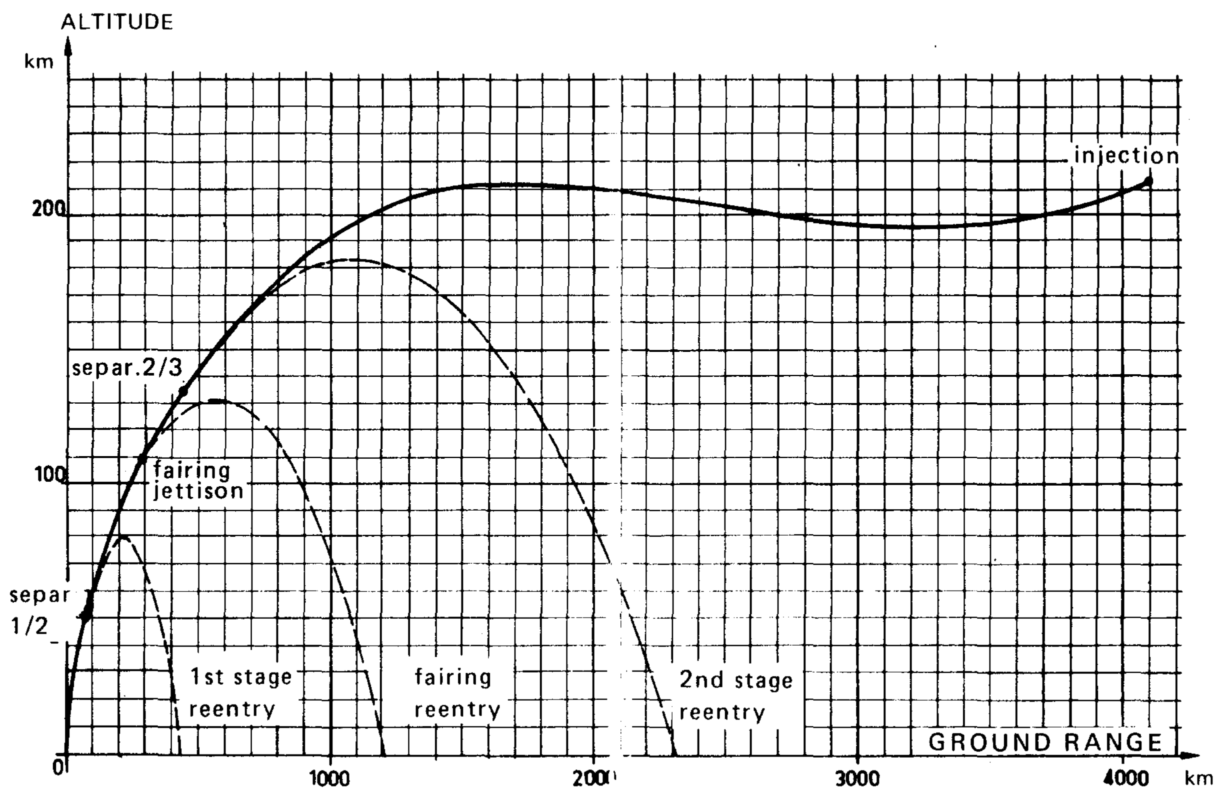
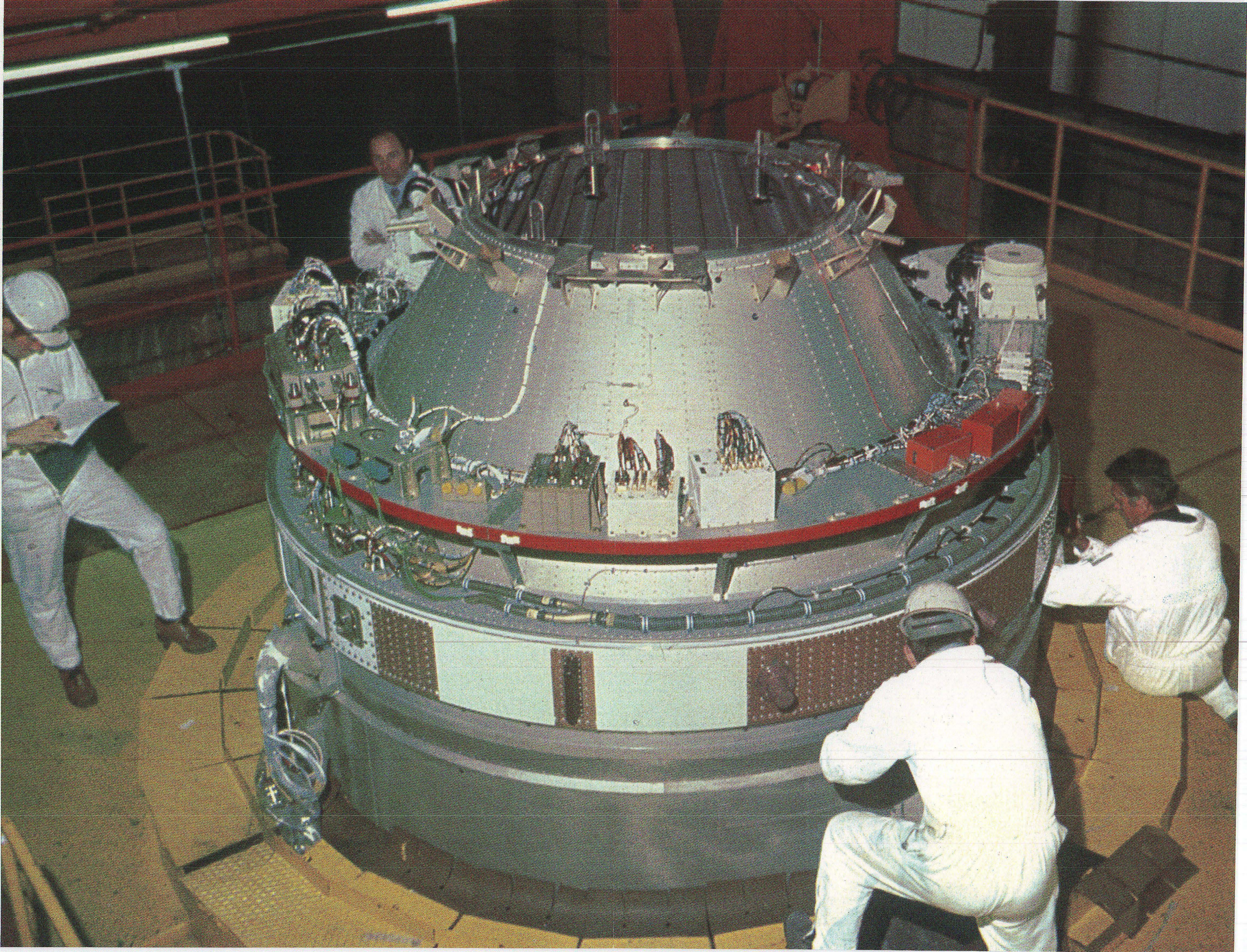


FIG. 2.18 TYPICAL FLIGHT PROFILE – GEOSTATIONARY MISSION



Launch vehicle / payload interface chapter 3

3.1. Mechanical interface

The payload is mounted on the top of the launch vehicle via an adaptor, connected to the vehicle equipment bay (VEB). It is protected by a fairing, enclosing the payload during the atmospheric part of the flight.

The mounting plane for the payload is horizontal to within $\pm$ 2 mrd.

Adaptors 1194, 937, 1497 and the Sylda dual-launch adaptor form part of the launch vehicle. Other adaptors can be used, by arrangement, at the user's request.

A key to the symbols used in the figures showing mechanical interfaces is given in figure 3.18.

3.1.1. Launch vehicle / payload adaptors

3.1.1.1. Adaptor 1194 (figs. 3.1 to 3.6c)

In this version, the adaptor is a structure in the form of a truncated cone with an alignment diameter of 1194 mm in the payload separation plane. It is attached to the VEB by means of a bolted connector frame. It enables payload separation.

The payload rests on the forward frame of the adaptor and is secured by a clampband. The latter consists of a metal strip which holds in place a series of clamps hooked on to the payload and adaptor frames. The preloading of the band ranges between 15 000 and 28 000 N. At separation, the band is severed in three places by three explosive charges mounted on the adaptor, the pieces remaining captive to the adaptor. The satellite is forced away from the launch vehicle by six actuators forming part of the vehicle and bearing on the payload rear frame. At integration, the actuators are restrained by bolts, so that no force is applied to the payload during assembly. Once the clampband is fitted and the bolts are withdrawn, each actuator exerts a force of less than 900 N on the rear payload frame.

Orientation of the payload with respect to the launch vehicle is by an indexing pin on the Z axis (fig. 3.3a).

On the adaptor forward frame, two areas are reserved as a bearing surface for the payload-separation sensors, which indicate the actual instant of separation (fig. 3.3b).

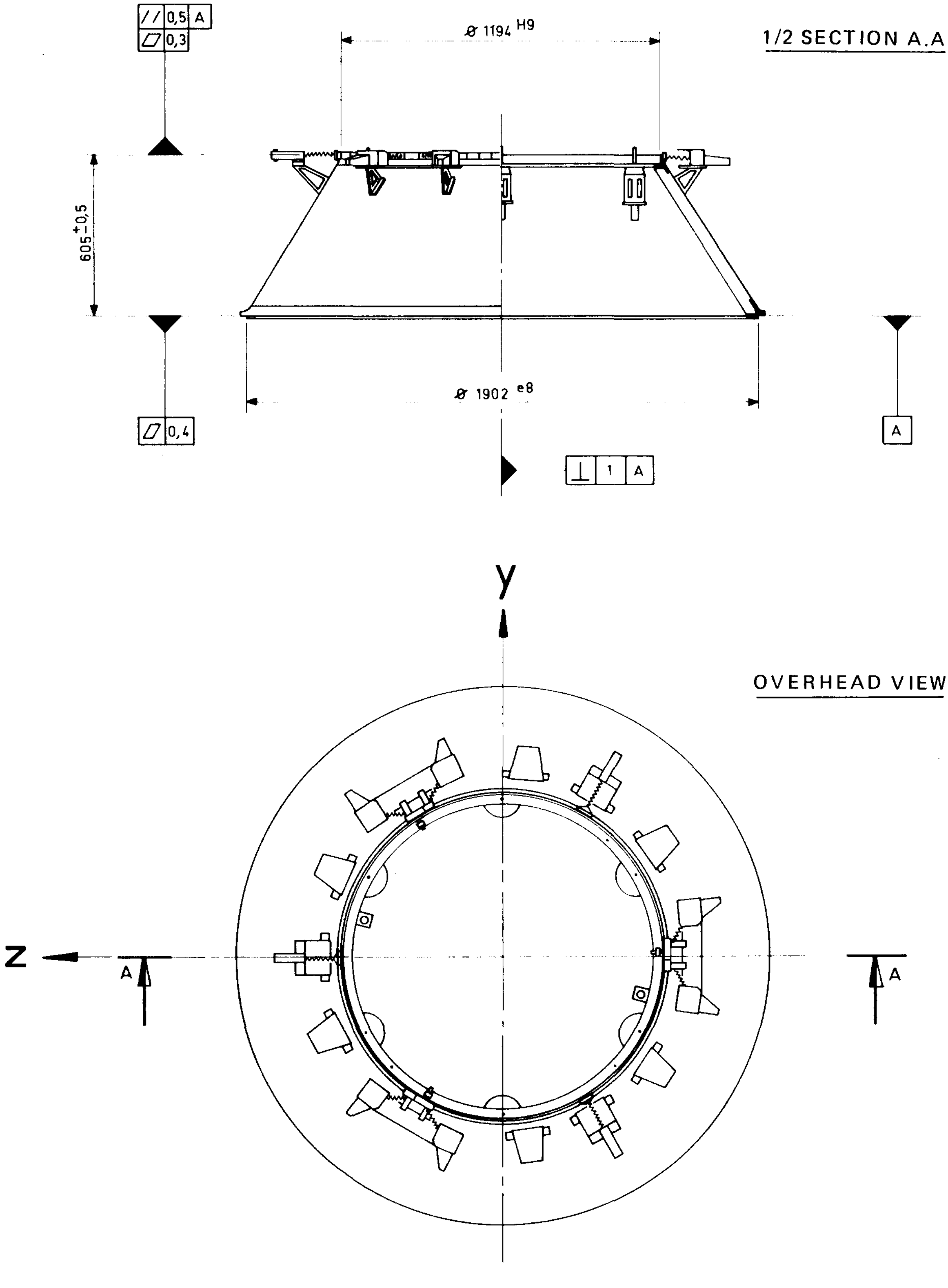


FIG. 3.1 PAYLOAD ADAPTOR 1194

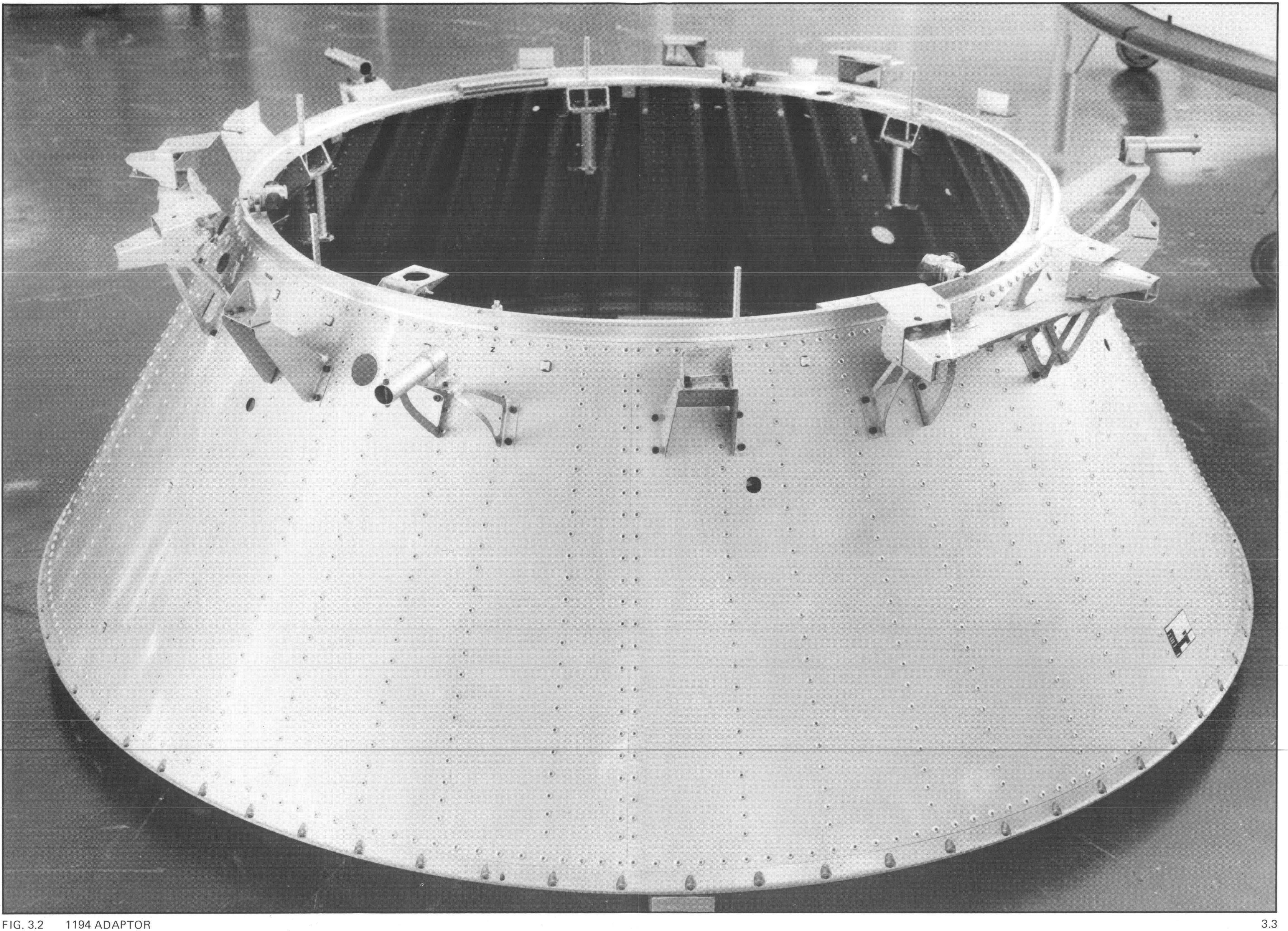


FIG. 3.2 1194 ADAPTOR

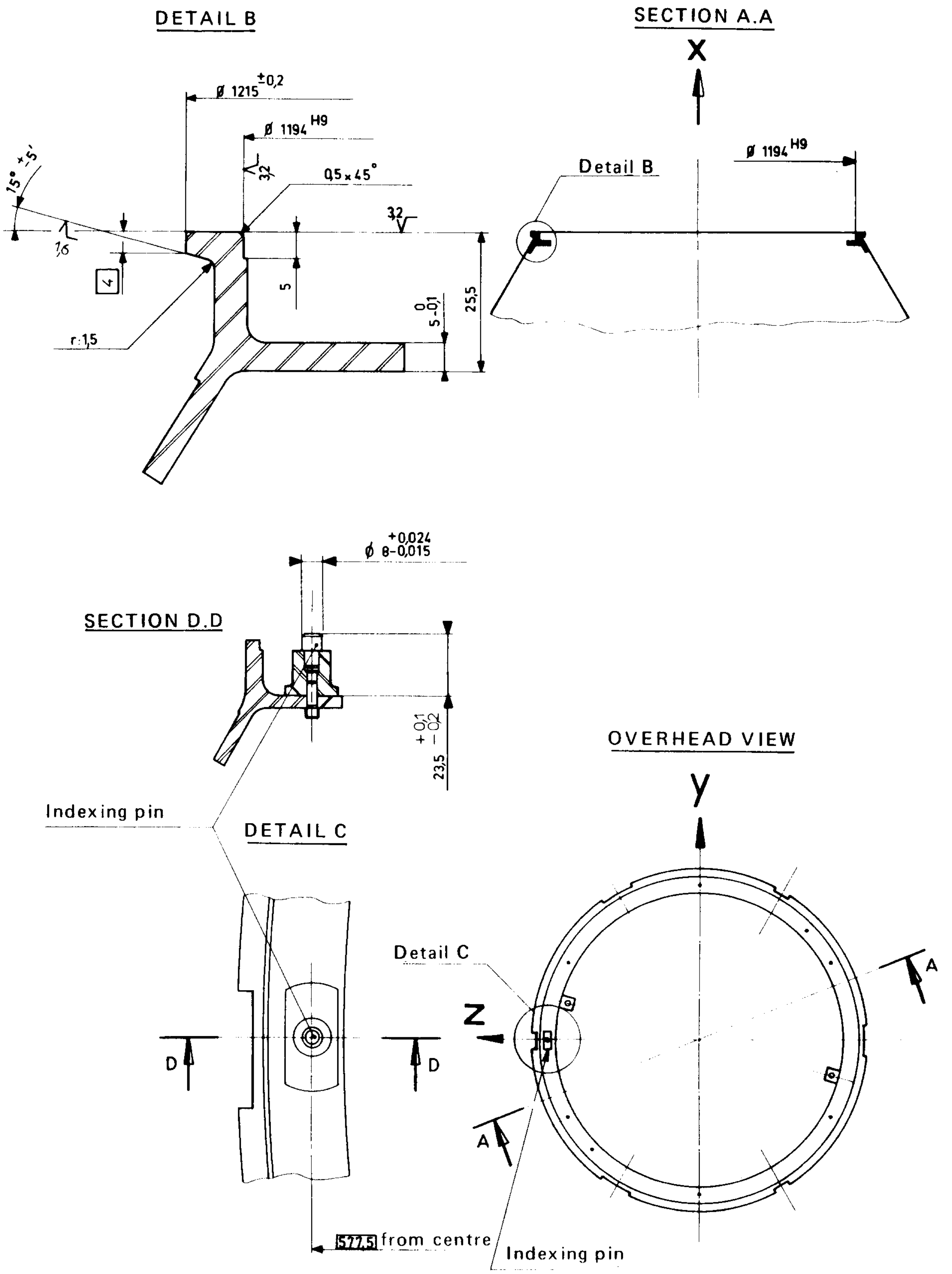


FIG. 3.3a 1194 PAYLOAD ADAPTOR FORWARD FRAME

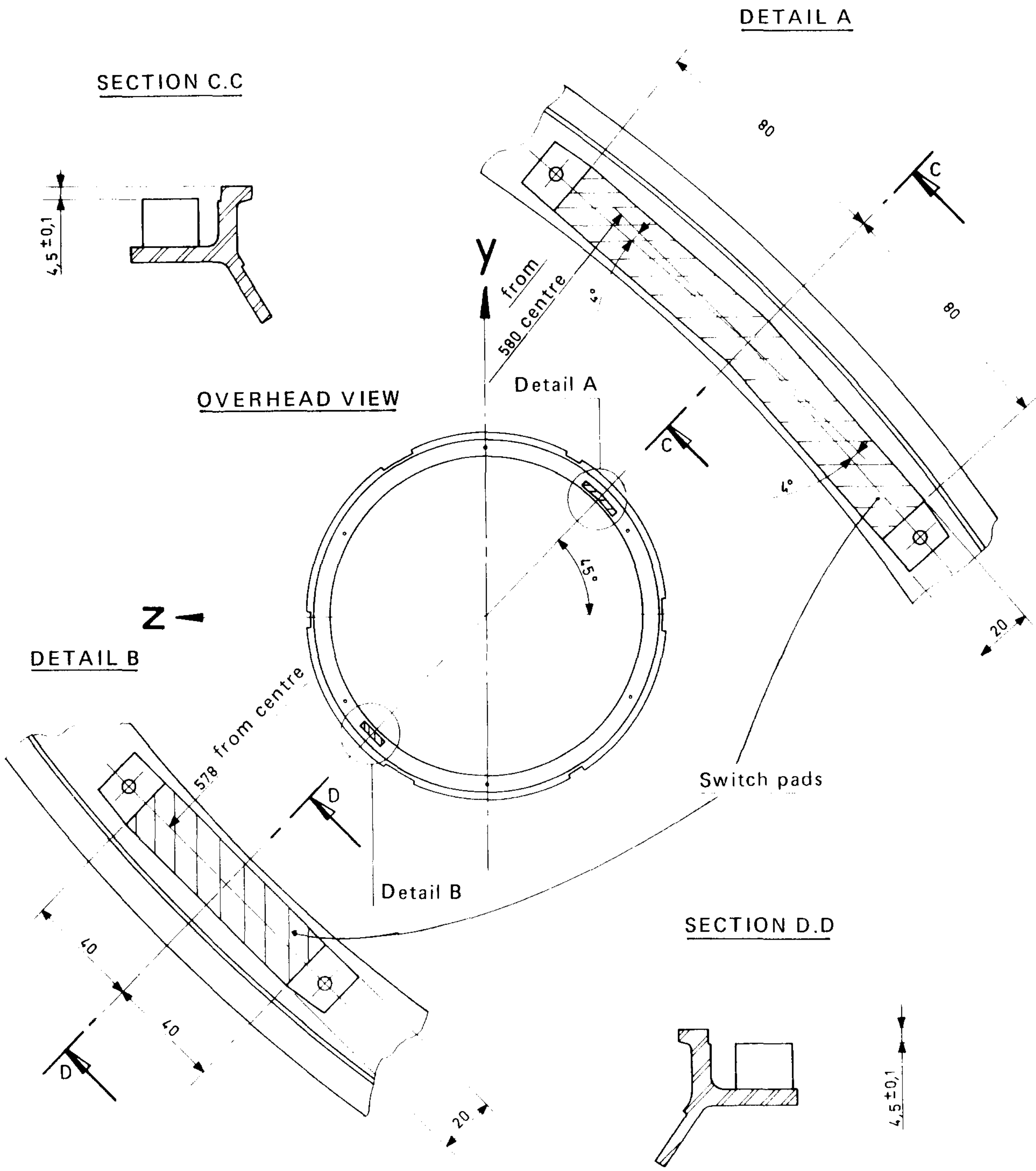


FIG.3.3b . 1194 PAYLOAD ADAPTOR FORWARD FRAME

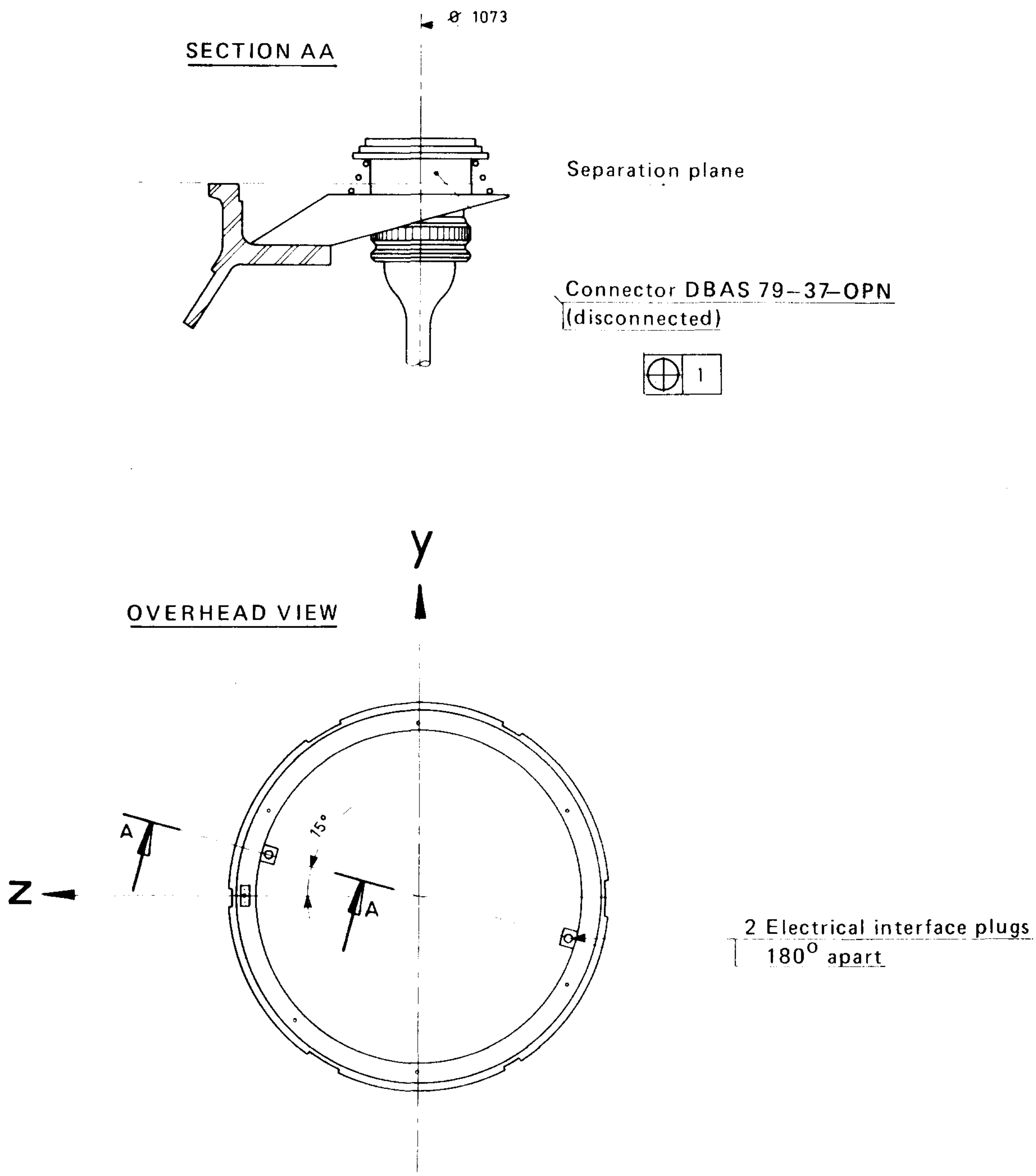


FIG.3.3c 1194 PAYLOAD ADAPTOR FORWARD FRAME

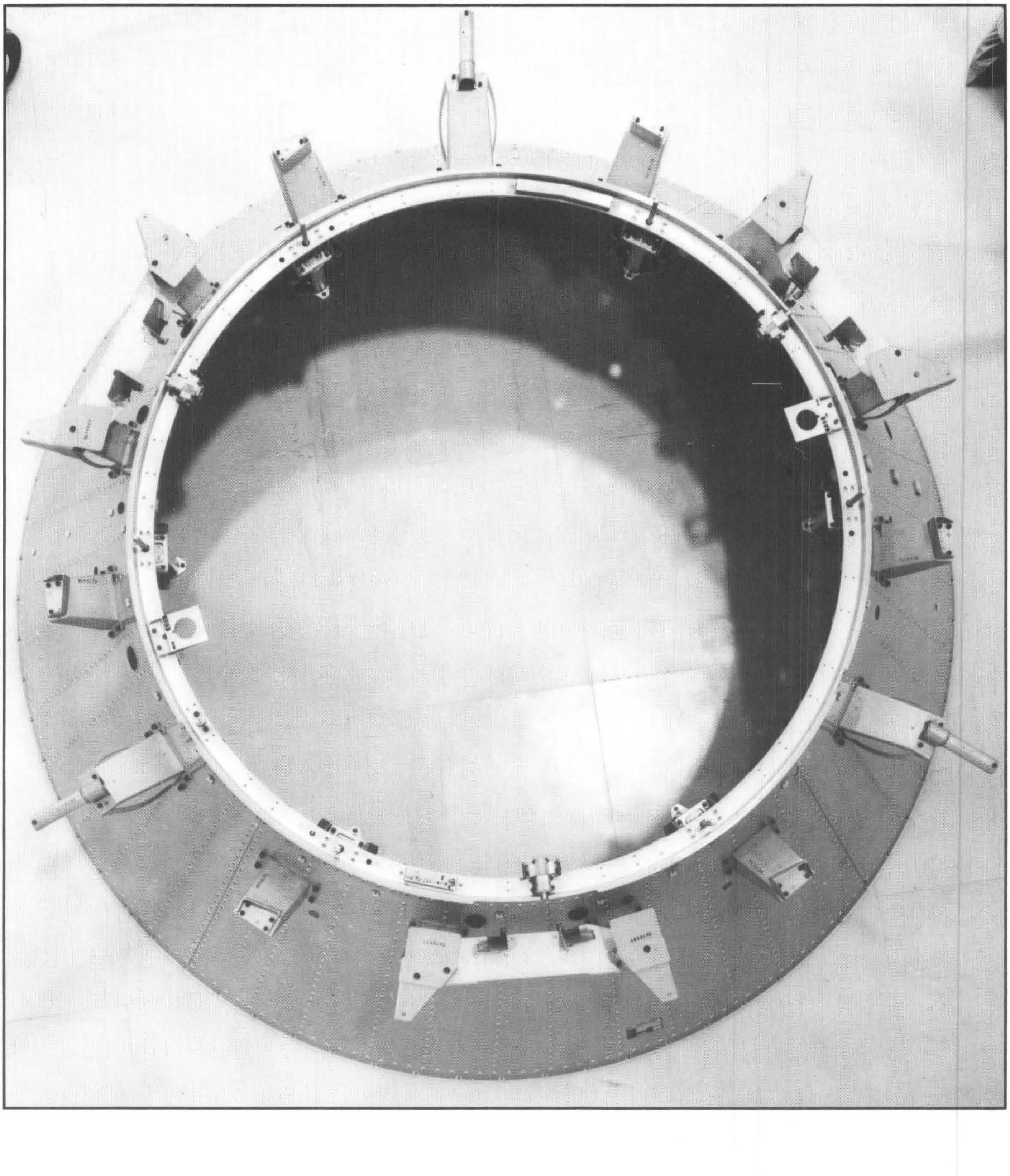


FIG. 3.4 1194 ADAPTOR – FRONT FRAME DETAIL

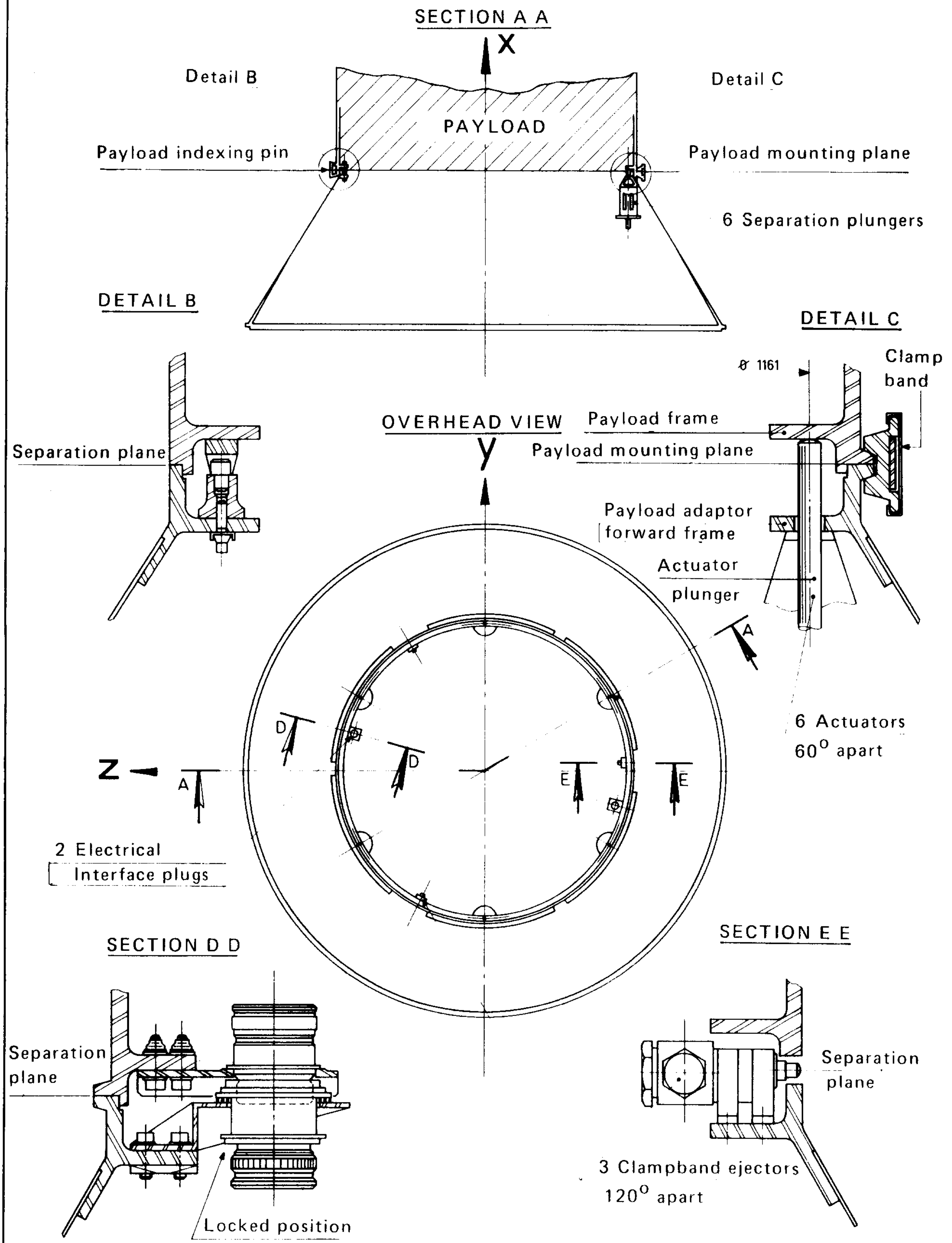


FIG 3.5 1194 ADAPTOR
PAYLOAD COMPOSITE

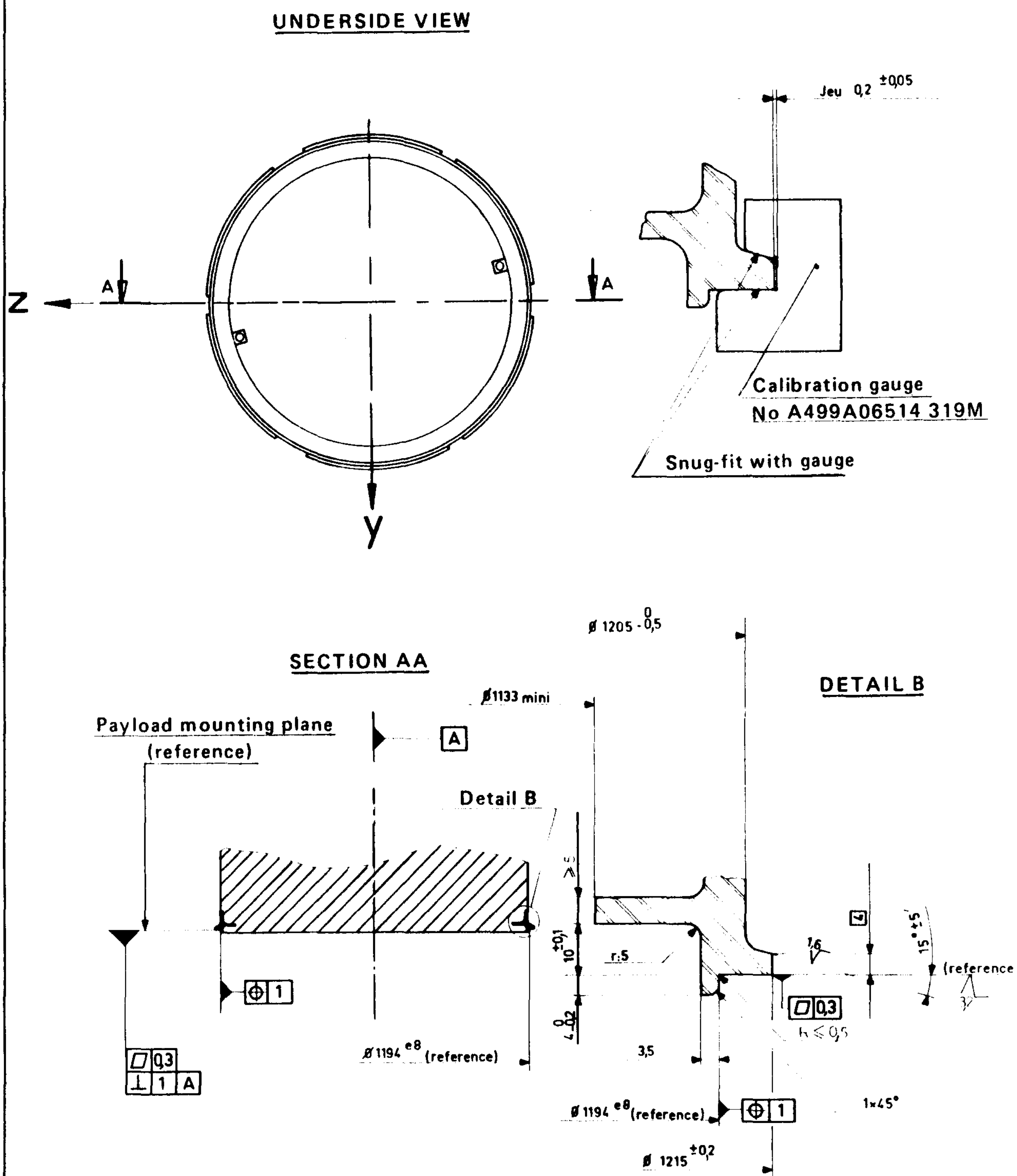
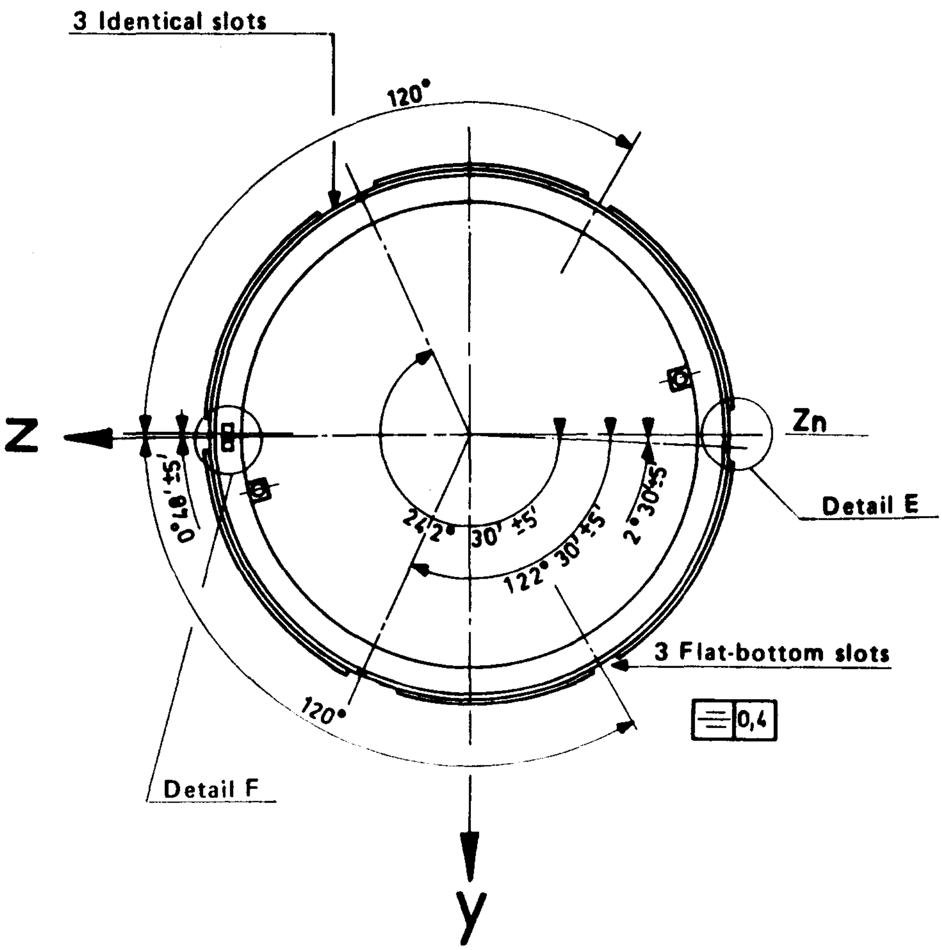


FIG. 3.6 a 1194 PAYLOAD ADAPTOR
REAR FRAME

UNDERSIDE VIEW



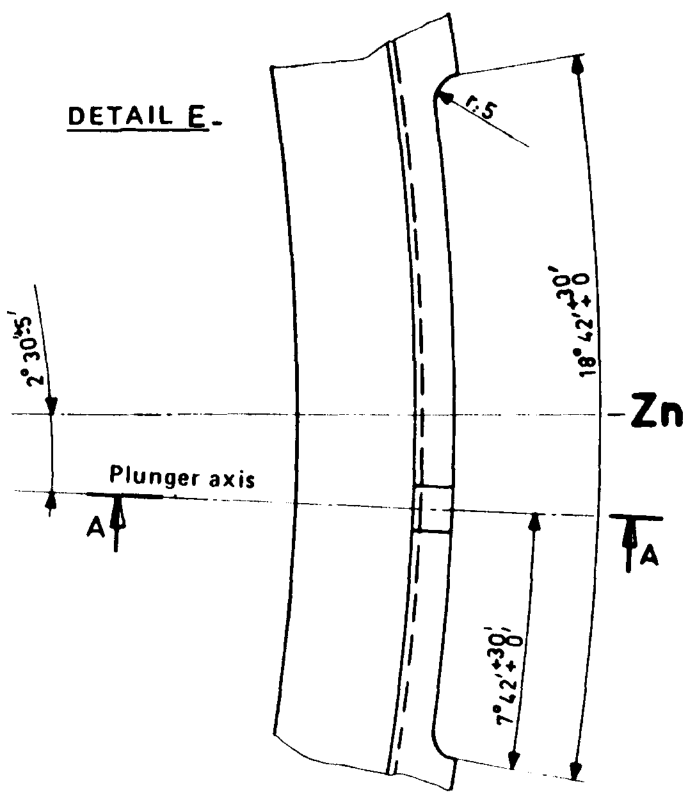


FIG. 3.6 b1 1194 PAYLOAD ADAPTOR
REAR FRAME

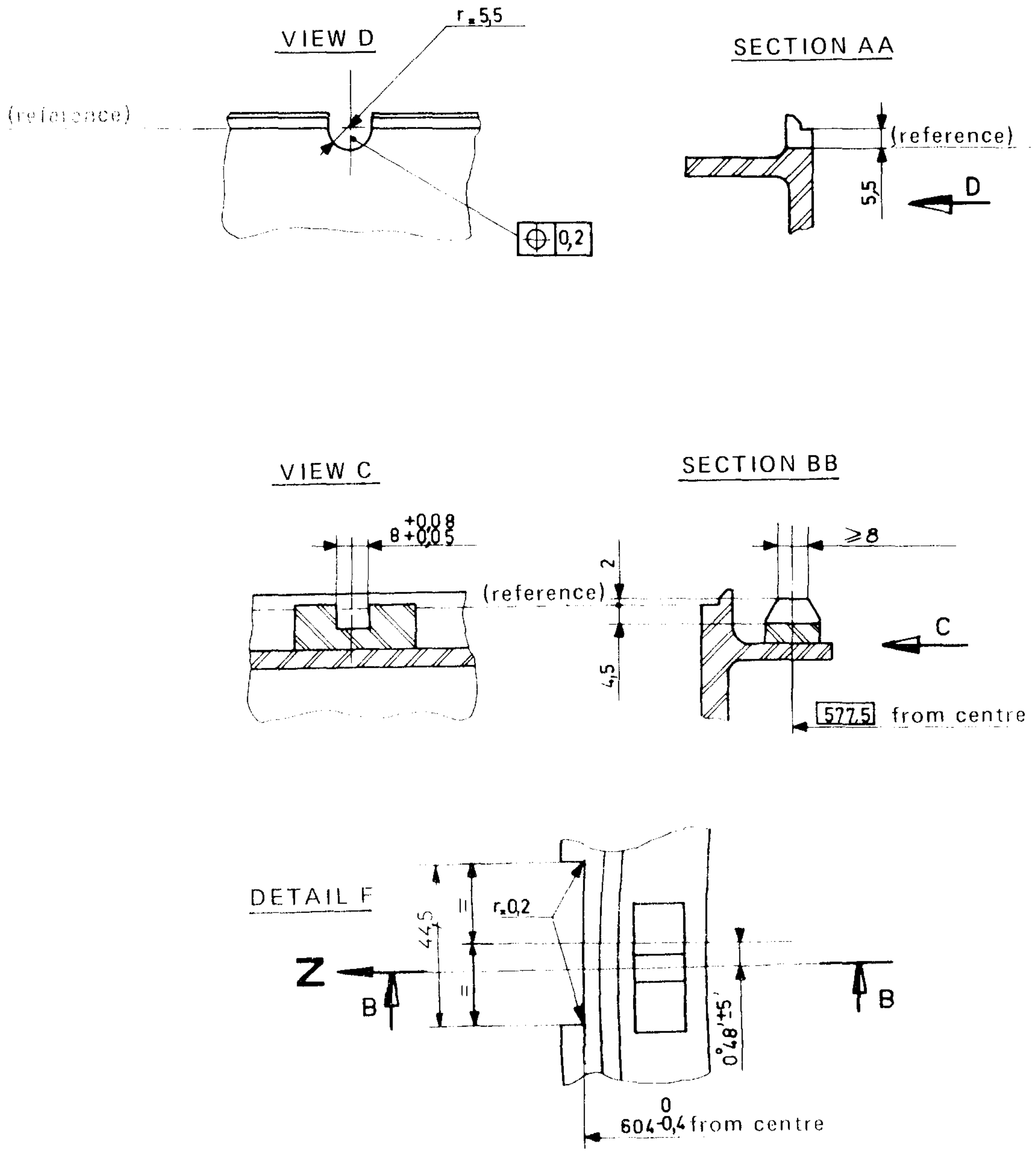
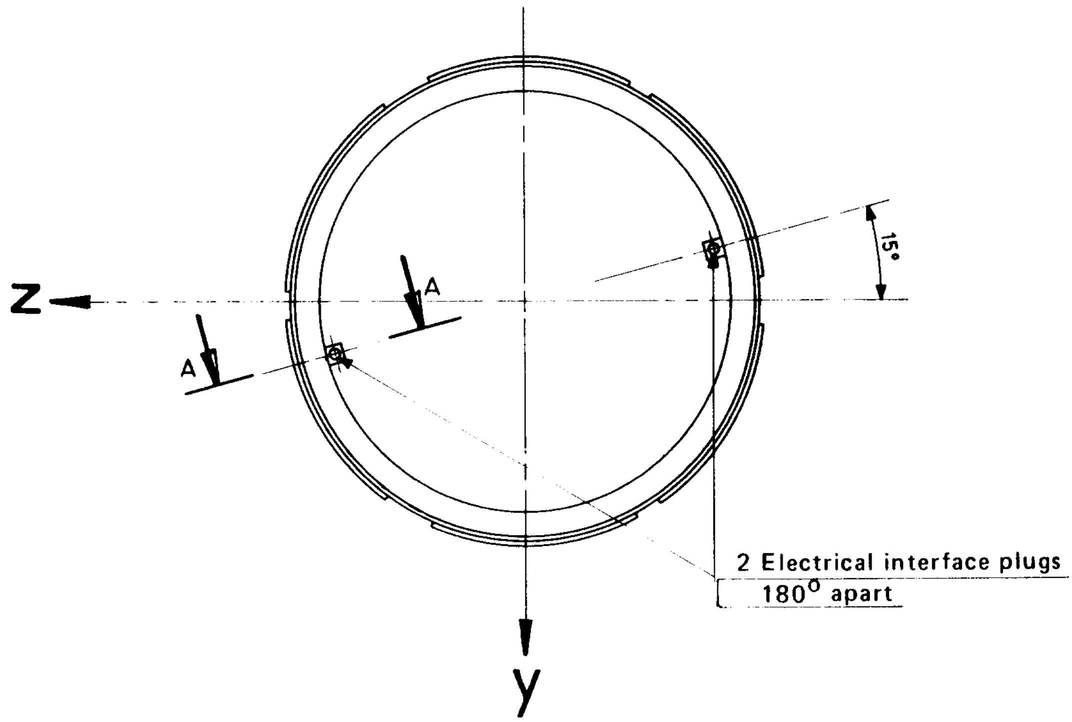


FIG.3.6 b2 1194 PAYLOAD ADAPTOR
REAR FRAME

UNDERSIDE VIEW



SECTION AA

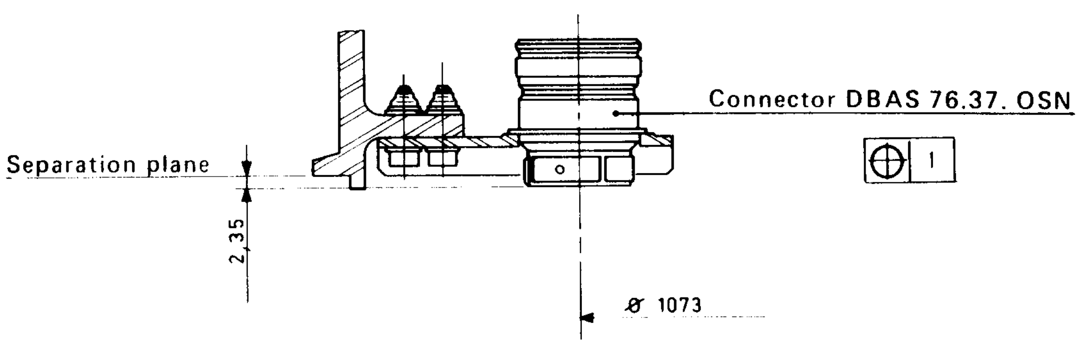


FIG.3.6c 1194 ADAPTOR
PAYLOAD REAR FRAME

The type 1194 adaptor has a mass of 43 kg.

Figure 3.6a gives the dimensions of the payload-frame butt. Apart from correct observance of these dimensions, this frame must be executed in an aluminium alloy having the following mechanical specifications : elastic limit 0.2 % $\geqslant 25.10^7$ Pa ; breaking stress $\geqslant 40.10^7$ Pa.

The clampband introduces an angular flexibility at the vehicle/payload interface of between 1 and 2.10^{-9} *rd/Nm*

3.1.1.2. Adaptor 937 (figs. 3.7 to 3.12c)

In this version, the payload adaptor is a structure in the form of a truncated cone, with an alignment diameter of 937 mm in the payload separation plane. It is attached to the VEB by a bolted connector frame, and also enables payload separation.

This adaptor can be used with payloads whose interfaces are compatible with the Thor-Delta (37" adaptor) and the Pam-D.

The type 937 adaptor has a mass of 46 kg.

3.1.1.3. Adaptor 1497 (figs. 3.13 to 3.16)

In this version, the payload adaptor is a structure in the form of a truncated cone, the connection with the payload being by means of bolts, arranged on a circle of 1497 mm. It is attached to the VEB by means of a bolted frame.

This adaptor can be used with payloads whose interfaces are compatible with the Centaur stage. It does not enable payload separation.

The type 1497 adaptor has a mass of 26 kg.

3.1.1.4. Sylda dual-launch adaptor

See chapter 7.

3.1.1.5. Other adaptors

Adaptors other than those described above can be used on Ariane. The constraints applicable to any such alternative adaptor are described in figure 3.17. This shows the volumes reserved for the bay equipment, and the 3rd-stage insulating diaphragm. The adaptor is bolted on to the VEB forward frame.

Orientation is by 2 indexing pins. Figure 3.17 also shows the dimensions for the adaptor rear frame. Apart from correct observance of these dimensions, this frame must be executed in aluminium alloy.

FIG. 3.7 937 ADAPTOR
(Will be published in a subsequent issue)

FIG. 3.8 937 ADAPTOR
(Will be published in a subsequent issue)

FIG. 3.9 937 ADAPTOR
(Will be published in a subsequent issue)

FIG. 3.10 937 ADAPTOR
(Will be published in a subsequent issue)

FIG. 3.11 937 ADAPTOR
(Will be published in a subsequent issue)

FIG. 3.12 937 ADAPTOR
(Will be published in a subsequent issue)

FIG. 3.13 1497 ADAPTOR
(Will be published in a subsequent issue)

FIG. 3.14 1497 ADAPTOR
(Will be published in a subsequent issue)

FIG. 3.15 1497 ADAPTOR
(Will be published in a subsequent issue)

FIG. 3.16 1497 ADAPTOR
(Will be published in a subsequent issue)

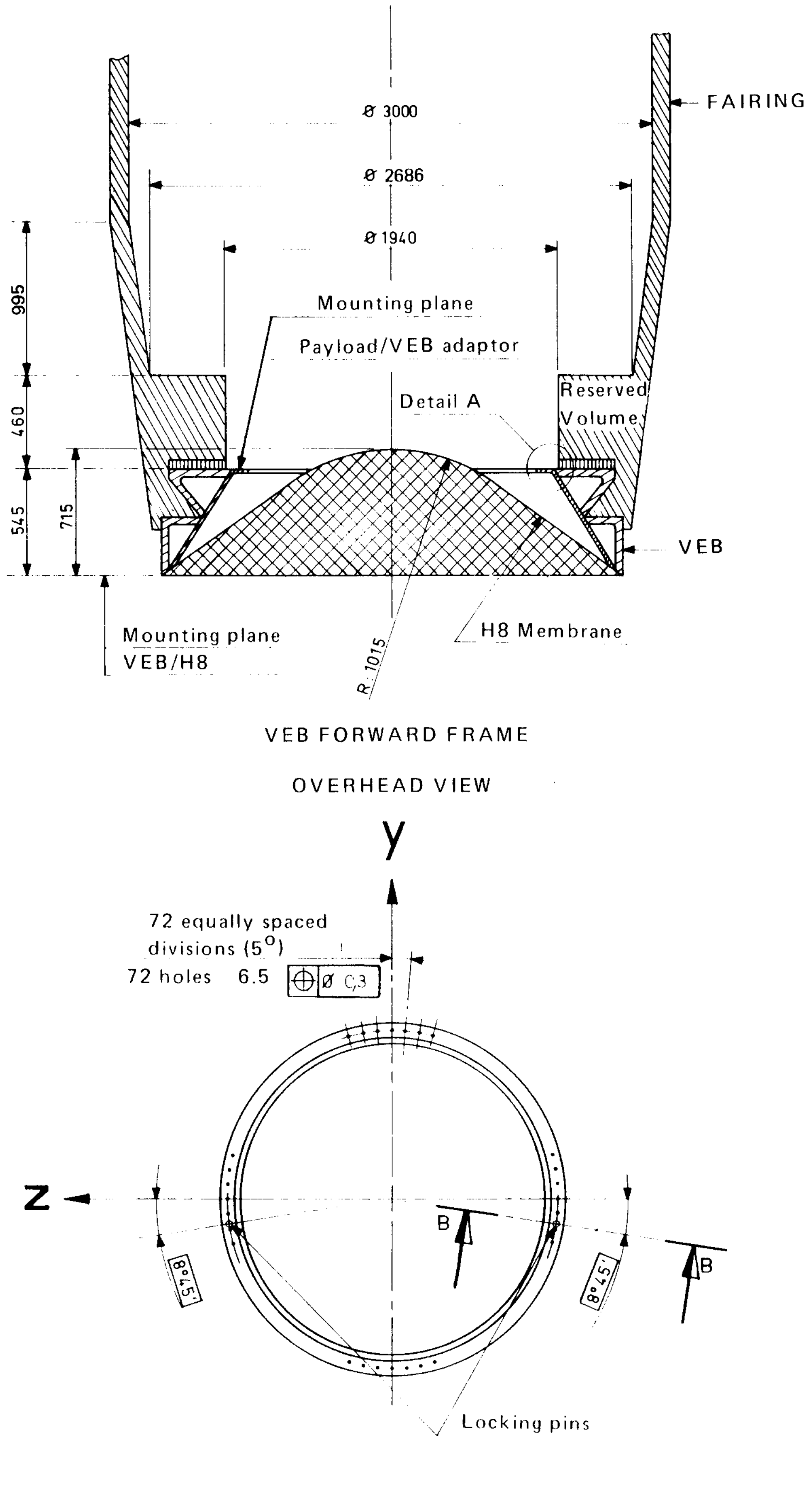


FIG.3.17 a – PAYLOAD ADAPTOR / VEB INTERFACE

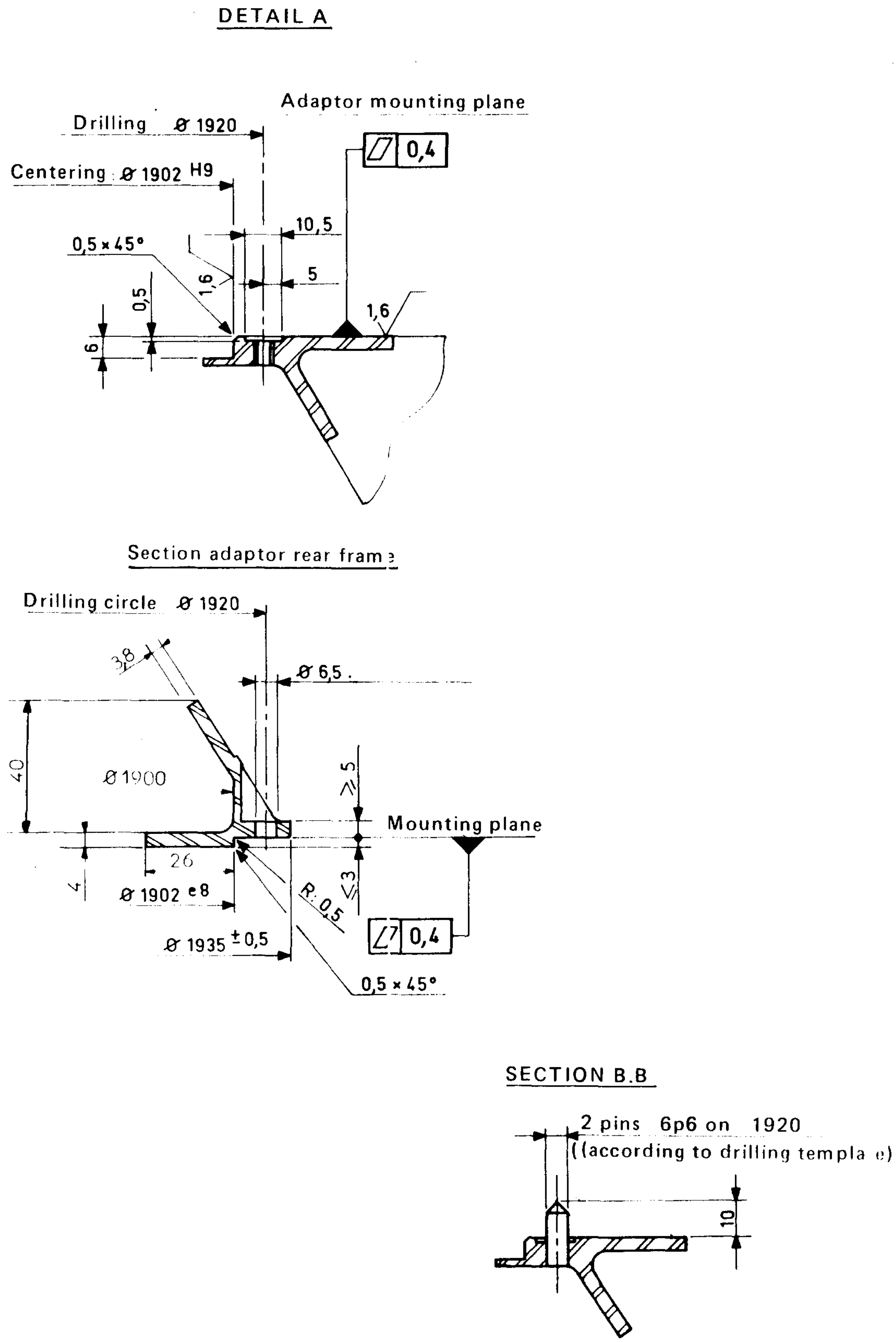
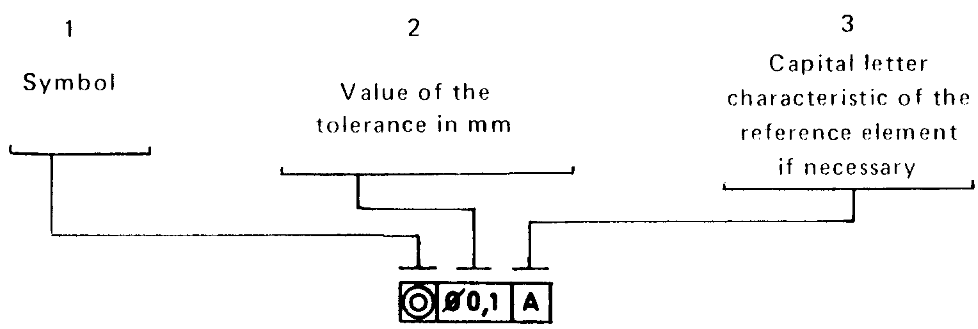


FIG.3.17 b – PAYLOAD ADAPTOR / VEB INTERFACE

Symbols for shape and position tolerances extracted from the R.E. Aero 772 75 norm, approved by BNAE.

TOLERANCE APPLICATIONS		SYMBOLS
SHAPE	Linear Planar Circular Cylindrical Any particular line Any particular surface	— ▱ ○ ⌭ ⌒ ⌓
ORIENTATION	Parallel Perpendicular Inclined	// ⊥ ∠
POSITION	Location of an element Concentric and coaxial Symmetrical	⌖ ◎ ⌯
	Backlash	↗

The frame is attached to the element subjected to the tolerance and features the necessary positions to specify the following indications.



EXAMPLE

FIG. 3.18 – EXPLANATION OF SYMBOLS

Note 1 : *Adaptor axial rigidity must be* $\geqslant$ *5.10*^{8} *N/m (the end frames being radially free).*

Note 2 : *Vents having a total surface area* $\geqslant$ *1520 mm*^{2} *must be provided in the adaptor skin.*

3.1.2. Nose fairing

3.1.2.1. Description

The Ariane nose fairing (see figs. 3.19 a, b and c) is a structure 8653 mm high, with an external diameter of 3200 mm, made up of two half-shells. It comprises three parts. The forward cone, surmounted by a hemispherical nose, is made of light alloy with cork protection. The cylindrical portion has a metal skin, stiffened by frames and stringers. It is 4 m long. The rear portion forms a boat-tail section whose diameter is reduced from 3200 mm to that of the VEB (2600 mm). It is made of phenolic resin honey-comb sections, reinforced with metal stringers. This portion has good radio-transparency characteristics (see para. 3.5.8.).

The fairing is attached to the VEB by its rear frame, which fits into a groove on the VEB forward frame, and is held in place during flight by a jettisonable clampband.

Fairing separation is obtained by means of a pyrotechnic cord, located close to the plane joining the two half-shells.

This cord shears the rivets connecting the two parts, and imparts a lateral impetus to the two half-fairings, driving them apart by a piston effect. The gases generated by the system are retained permanently inside an envelope, thus avoiding any contamination of the payload by the separation system.

3.1.2.2. Acoustic protection

Acoustic protection may be fitted inside the fairing.

It is made up of a number of " blankets " fitted to the elements of the cylindrical portion of the fairing : see figures 3.20a and b. The structure of a blanket is shown in detail in figure 3.20a. Specially shaped blankets may also be fitted to the fairing doors.

The acoustic blankets do not encroach on the useable volume defined in paragraph 3.1.2.3.

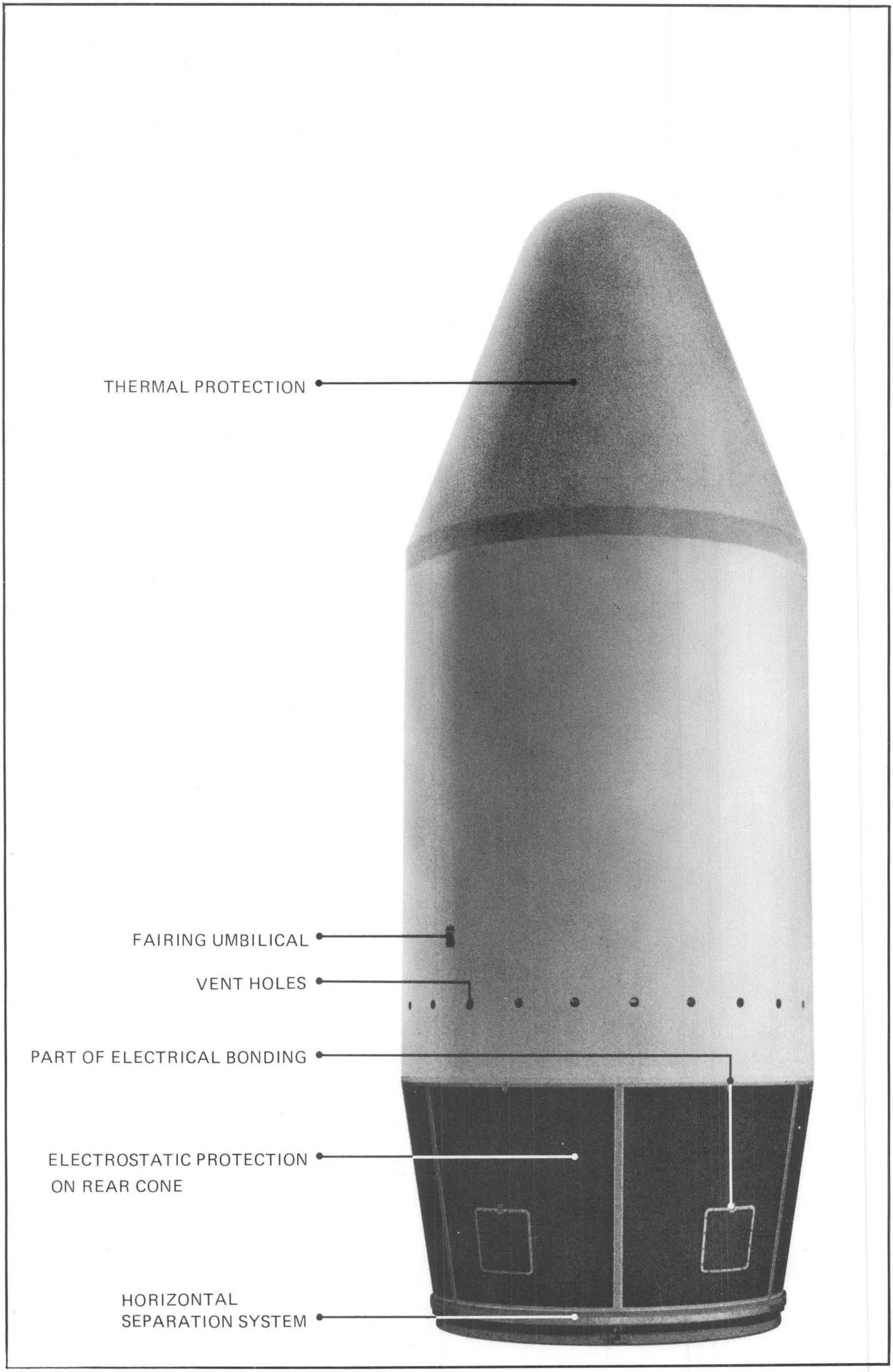


FIG. 3.19 a FAIRING

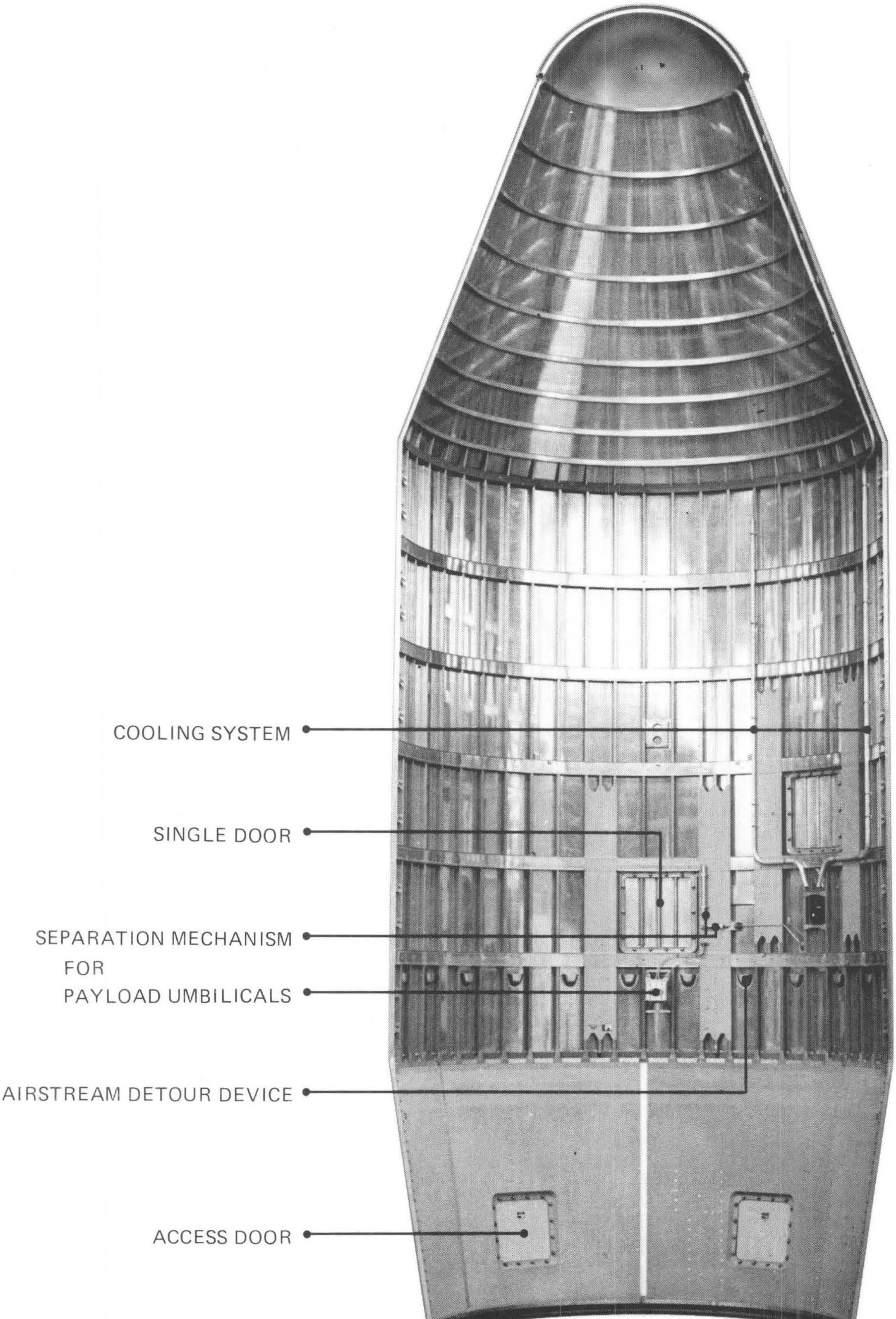


FIG. 3.19 b FAIRING – Z_N – HALF

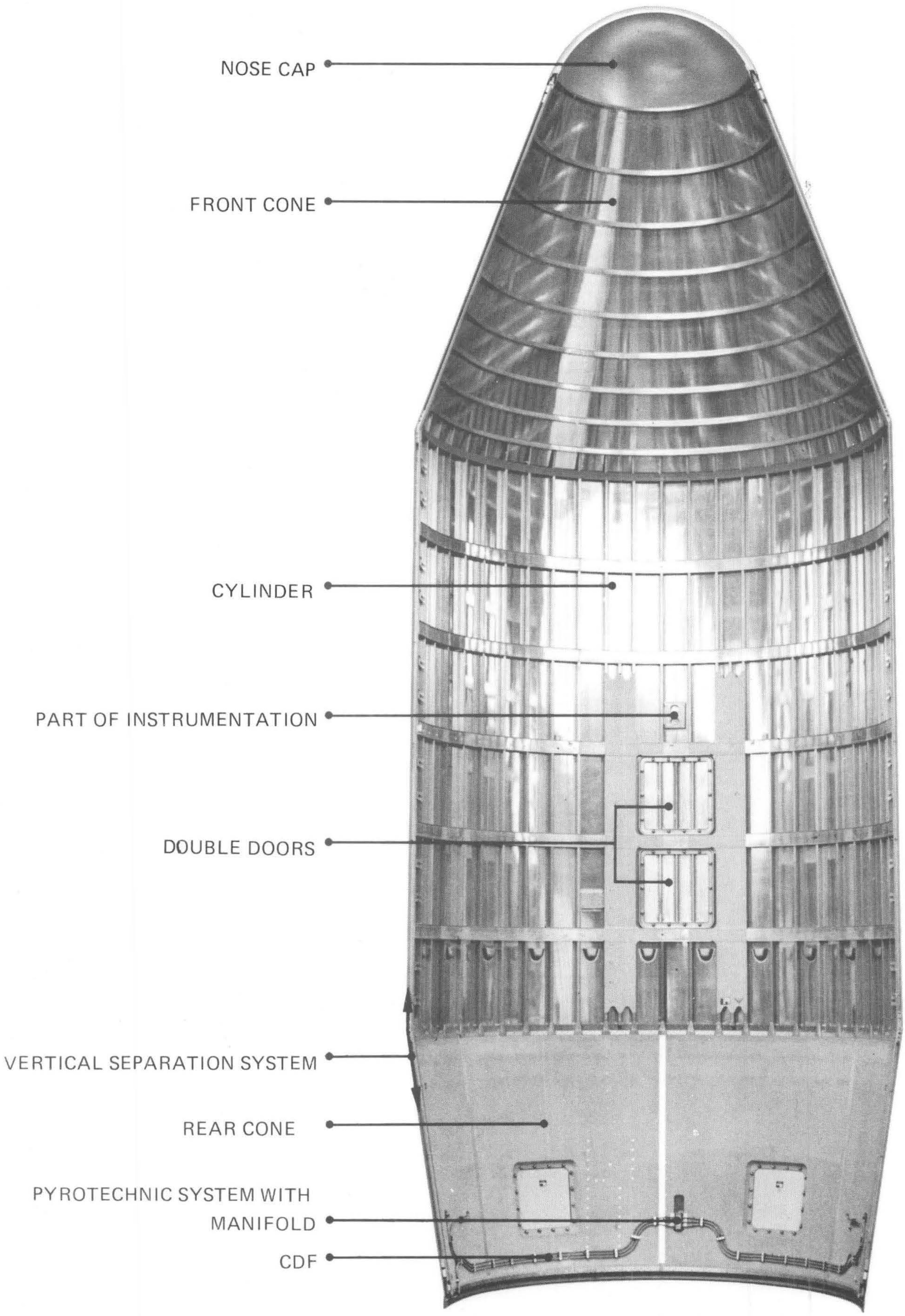
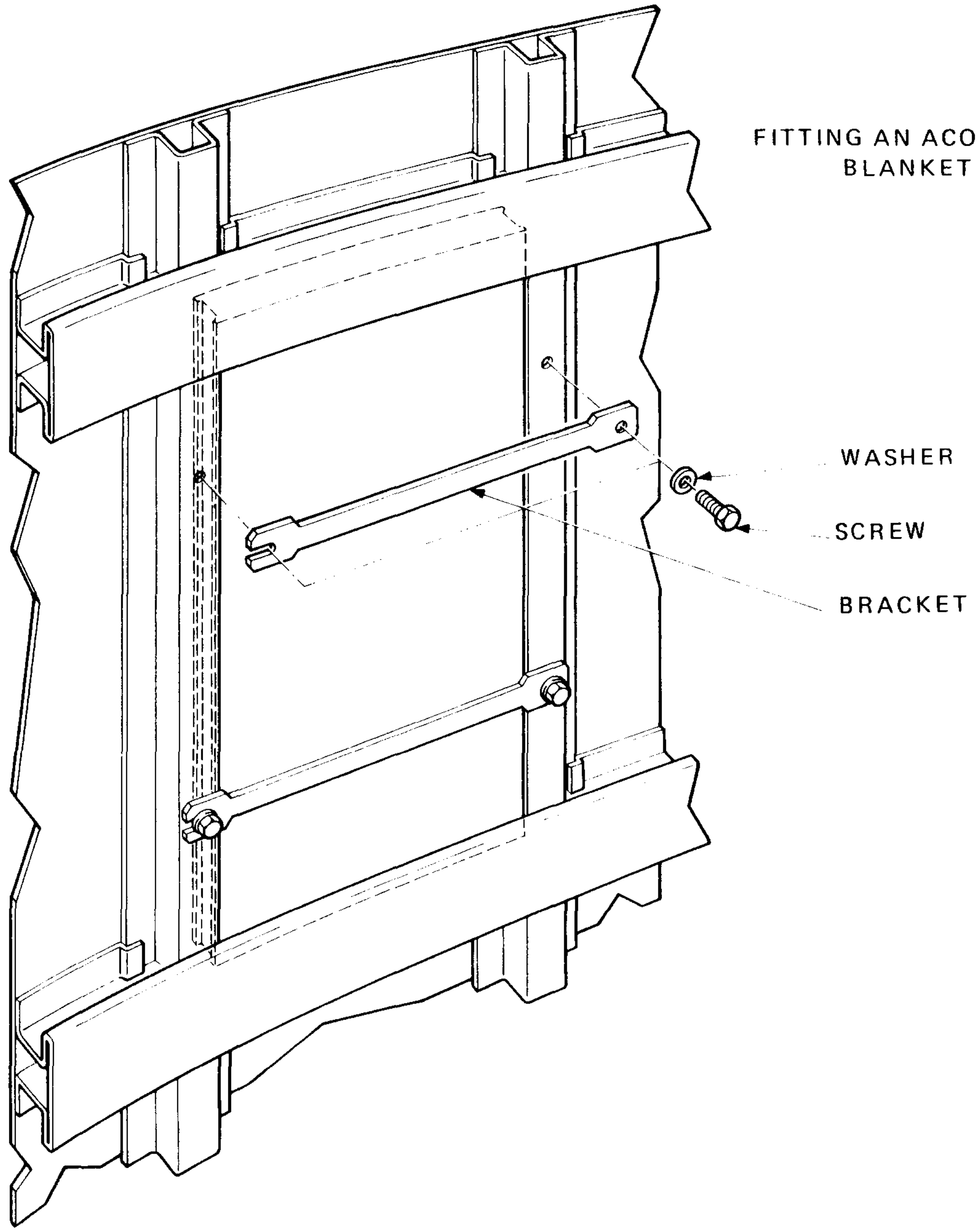


FIG. 3.19 c FAIRING – Z-HALF



ACOUSTIC BLANKET DETAIL

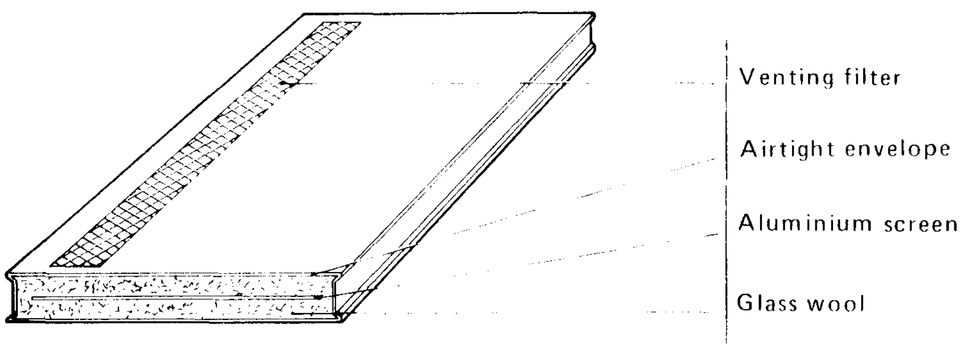


FIG. 3.20 a – ACOUSTIC BLANKET

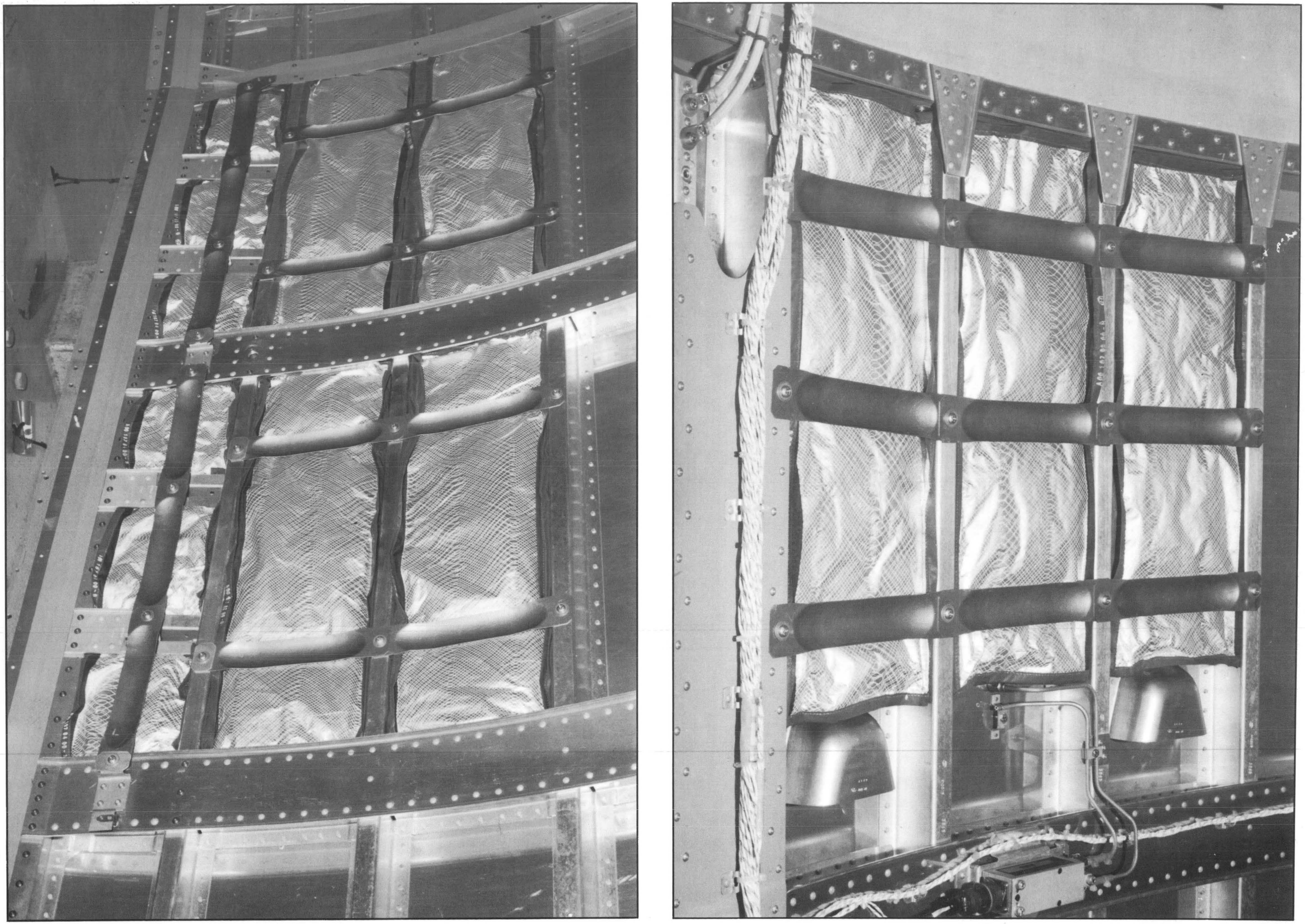


FIG. 3.20 b FAIRING WITH ACOUSTIC BLANKET

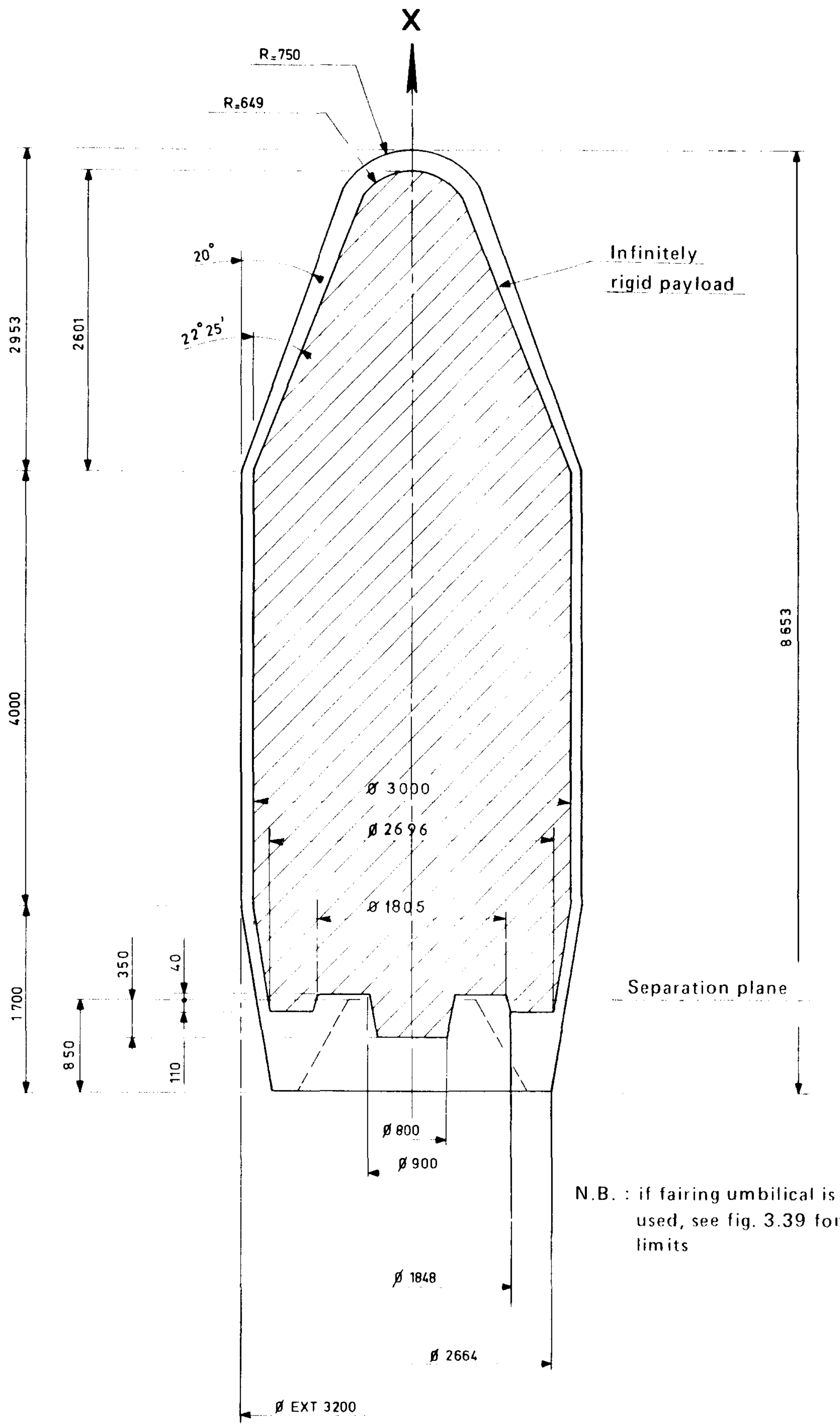


FIG. 3.21 a USABLE VOLUME UNDER FAIRING
ADAPTOR 1194

FIG. 3.21 b USABLE VOLUME UNDER FAIRING
937 ADAPTOR
(Will be published in a subsequent issue)

FIG. 3.21 c USABLE VOLUME UNDER FAIRING ADAPTOR 1497
(Will be published in a subsequent issue)

3.1.2.3. Useable volume

The free volume inside the fairing available to the payload, known as the " useable volume ", is shown in figures 3.21a, b and c (the useable volume for non-standard adaptors is shown in fig. 3.17). This volume constitutes the limits that the payload must not exceed, having regard to manufacturing tolerances and payload distortions due to static and dynamic loads in flight.

This volume assumes an infinitely rigid payload, but allowance has been made for the flexibility of the fairing and of the payload adaptor under the static and dynamic loads encountered in flight.

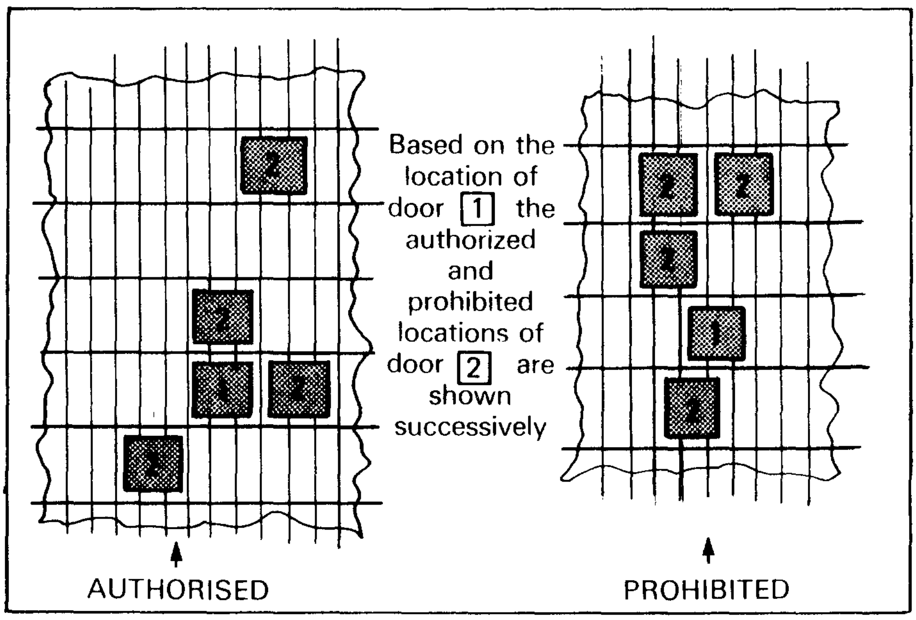
Figure 3.21a also shows, as an example, the useable volume for a payload having a first transverse frequency of 10 Hz.

Where payload flexibility is greater, or where elements of the payload are critical, the compatibility of the payload dimensions with the useable volume will be studied in greater depth, by coupled dynamic analysis (see chapter 6).

If used, the umbilical-plug extraction system inside the fairing encroaches on this volume (see para. 3.5.2.2.).

3.1.2.4. Payload accessibility

Figures 3.22a and b show a projection of the interior of the fairing. They show which areas are authorized, subject to negotiation, or prohibited for location of the fairing doors. The standard version includes two metal doors per half-fairing, with an effective aperture 487 mm high and 458 mm wide (equivalent to 3 cylindrical elements). The height of each door is demarcated by 2 consecutive frames, and its width by 2 stringers. Rules covering adjacent door proximity are illustrated in the following diagrams :



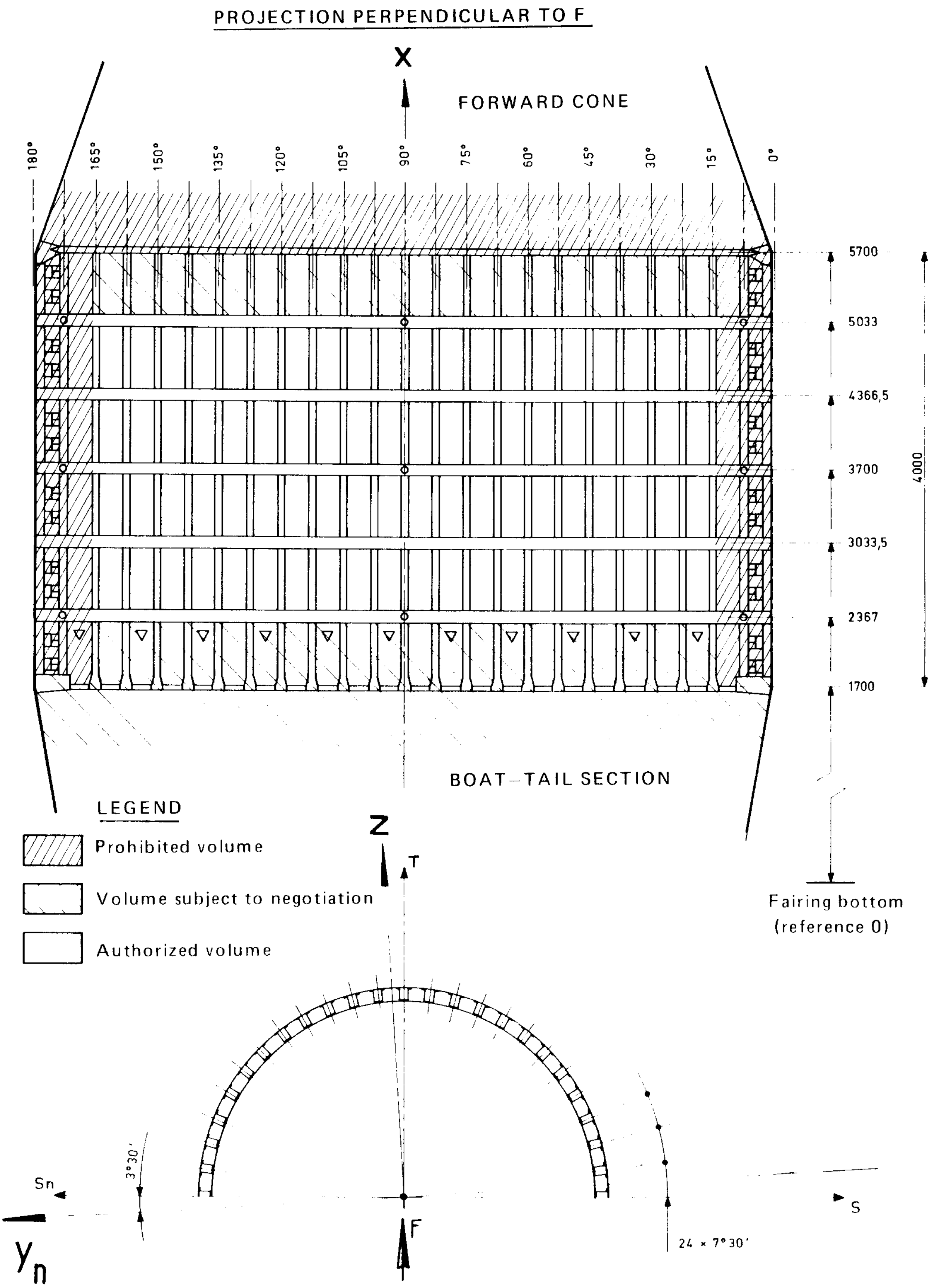


FIG. 3.22 a 1/2 FAIRING Z

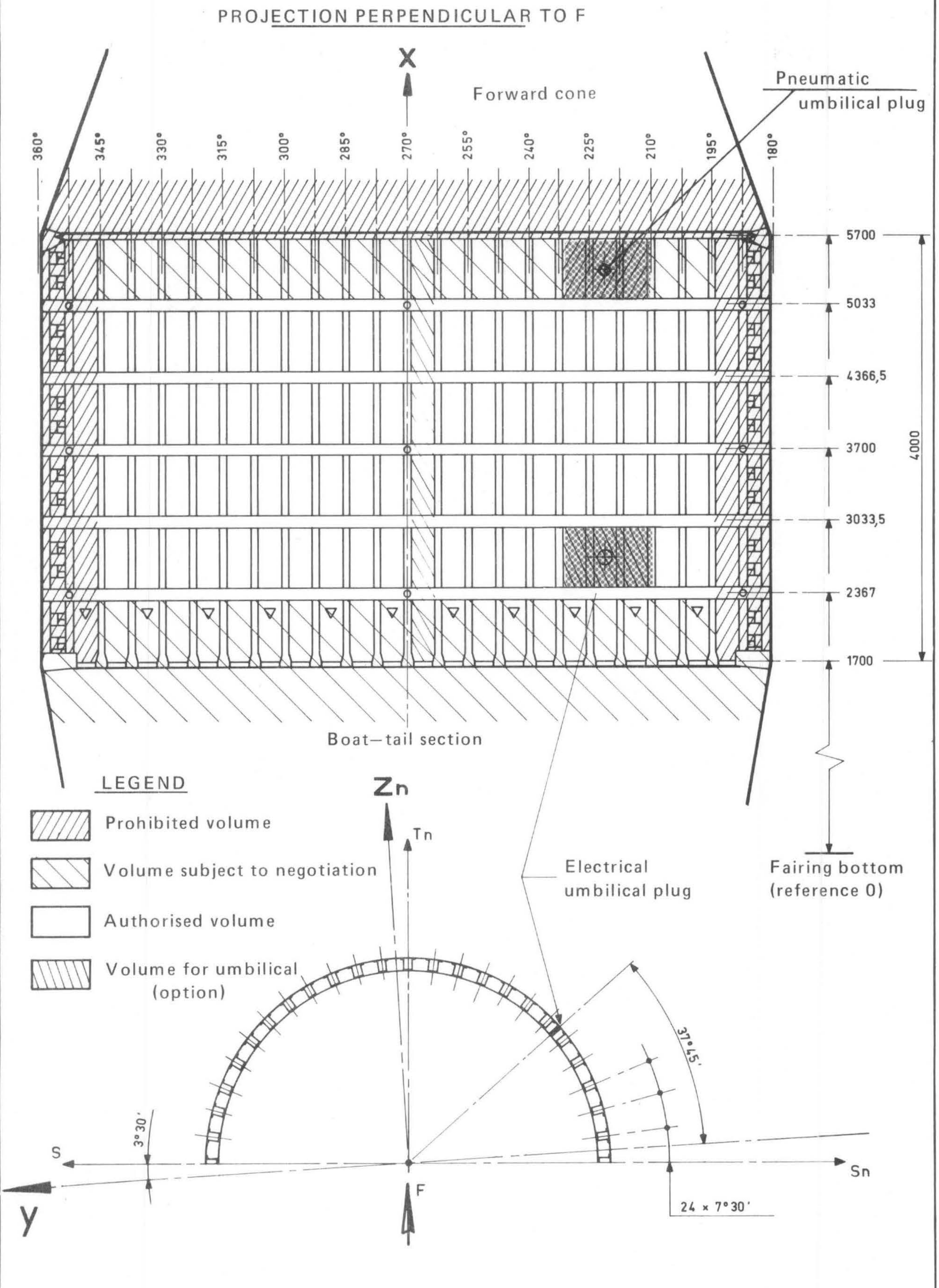
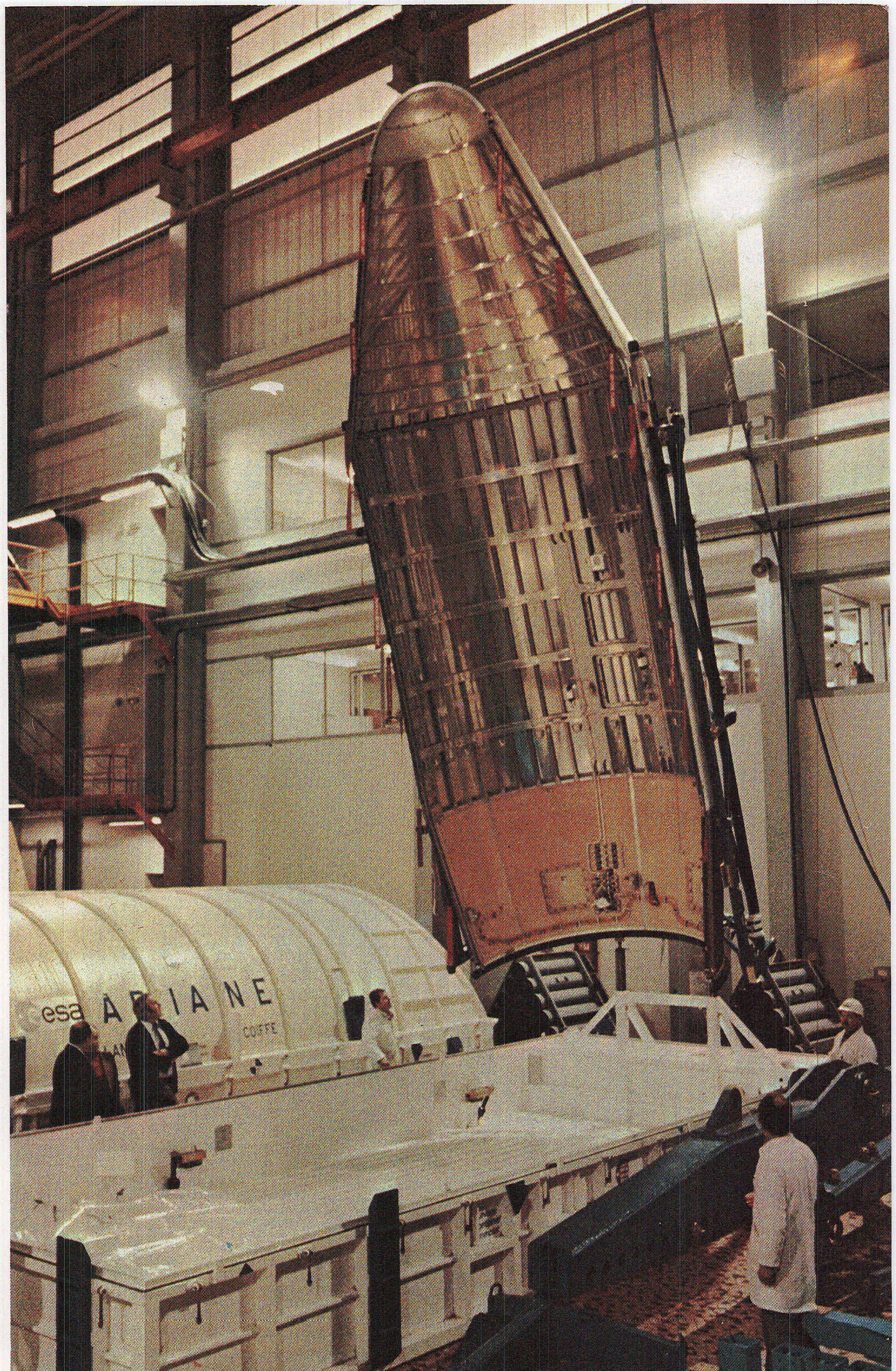


FIG.322 b 1/2 FAIRING Z_n


esa ARIANE
COIFFE

ARIANE shroud
lifting one half-shroud out of its transport container
(SIL, Les Mureaux)

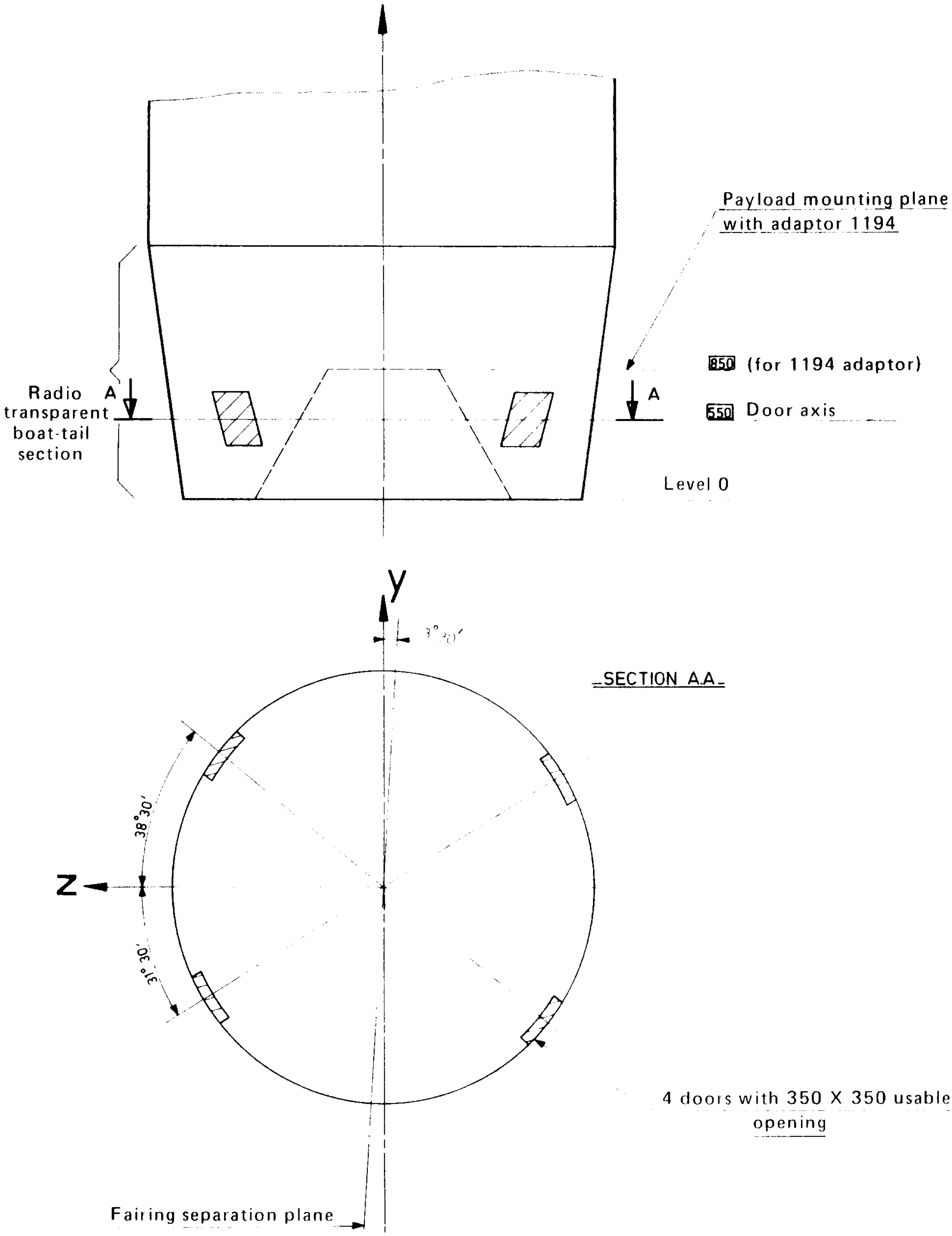


FIG. 3.22 c ACCESS AT FAIRING BOAT–TAIL SECTION

No access doors are included in the forward cone. In the boat-tail section (fig. 3.22c), four radio-transparent doors give access to the equipment in the VEB.

They may also be used to provide access to the payload. If requested, the fitting of an additional door in this section can be considered.

3.2. Mass - Alignment - Inertias

3.2.1. Payload mass

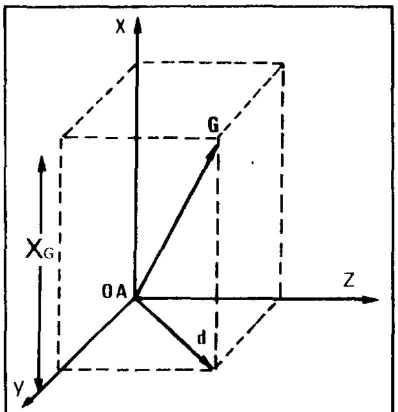
The total payload mass in launch configuration, deduced by weighing when the payload is ready for assembly with the launch vehicle, must not exceed by more than 2 per thousand the maximum value quoted for the final version of the launch-vehicle flight plan. This value must be notified at latest 8 months ahead of launch, with a maximum tolerance of $\pm$ 1 % (see chapter 6).

3.2.2. Payload alignment

In the three-axis system OA, X, Y, Z parallel to the vehicle reference axes passing through the centre of the launch-vehicle/payload separation plane, the centre of gravity of the payload must be at a height X_G and a distance d from the axis OAX. If Mcu is the mass of the payload, the user must design the latter so that :

$d.Mcu \leq$ 150 m.kg for non-spinning satellites

$d \leq$ 0.010 m for spinning satellites



static unbalance

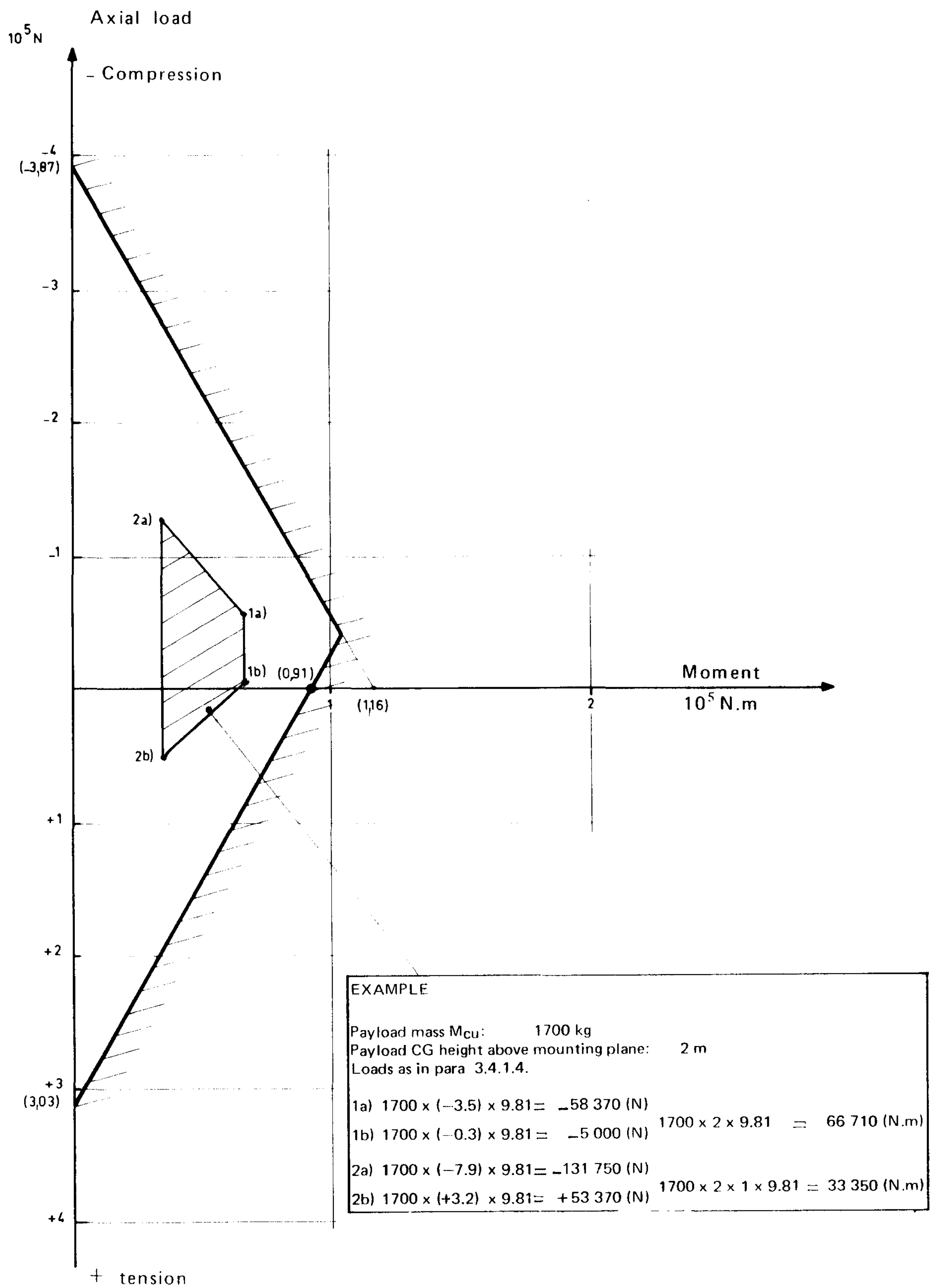


FIG. 3.23 a LIMIT LOADS 1194 ADAPTOR AT ADAPTOR/PAYLOAD INTERFACE

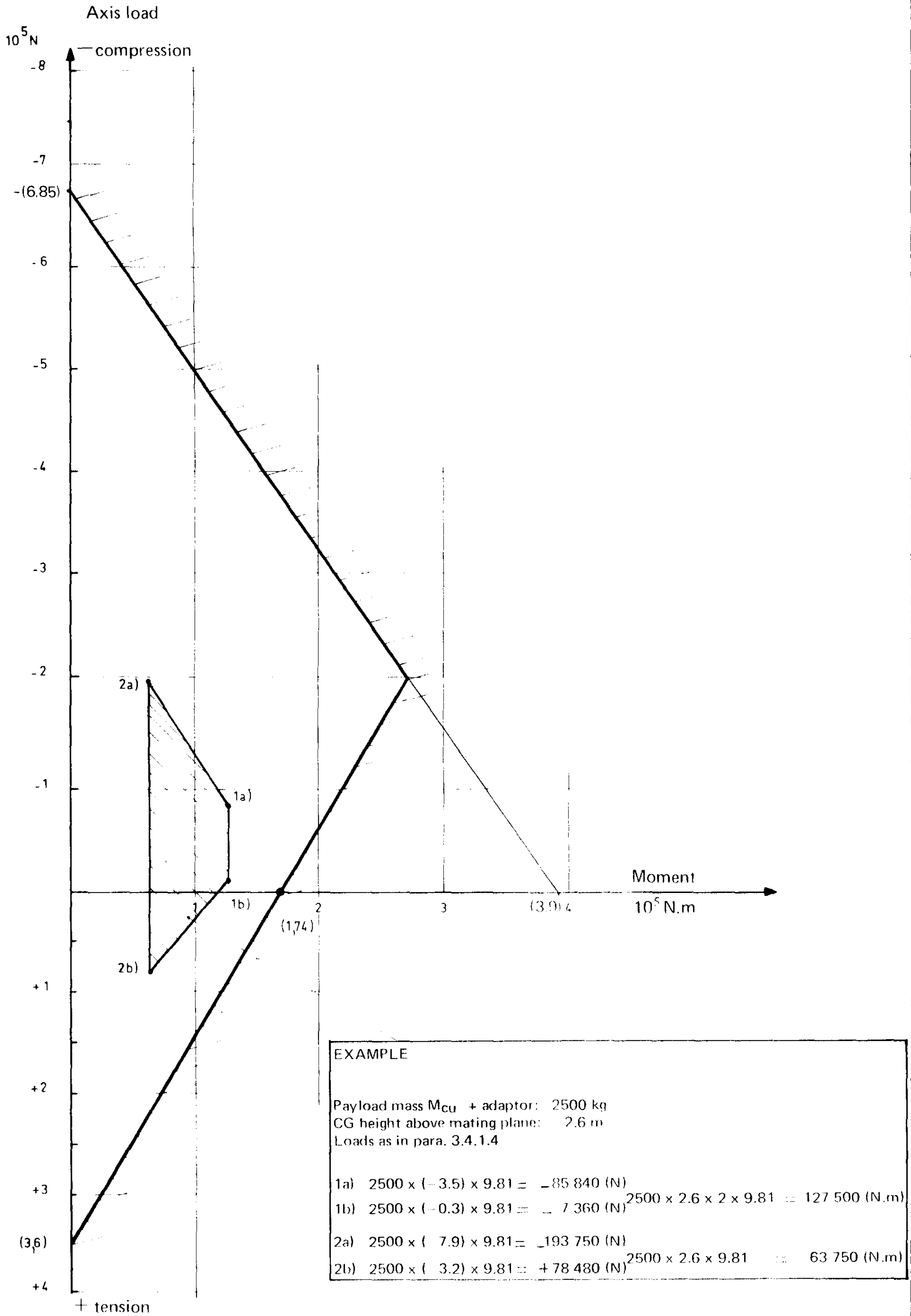


FIG. 3.23 b LIMIT LOADS ON LAUNCHER AT PAYLOAD ADAPTOR/VEB INTERFACE

Furthermore, the pair of values (Mcu, X_G) must be such that the forces on the launch-vehicle/payload interface (bending moment and normal force) remain within admissible limits, as defined below, using the quasi-static load values indicated in para. 3.4.1.4.2. :

figure 3.23a	with adaptor 1194
figure (TBD)	with adaptor 937
figure (TBD)	with adaptor 1497
figure 3.23b	with non-standard adaptor

If the payload cannot meet these specifications, the payload authority must request the Ariane authority to examine the repercussions on the launch vehicle.

3.2.3. Payload inertias

The permitted ranges of pitch, yaw and roll inertia values are wide, and no specification is therefore laid down in this manual. Once these values are known, however, they must be accurate to within 10 %.

If the payload is spun up at 10 rpm, the dynamic unbalance, corresponding to the angle between axis OAX and the main roll inertia axis, must be less that 1 degree.

3.3. Kinematic conditions at separation

Between 3rd-stage propulsion cut-off (H3) and payload separation (H4) commanded by the launch vehicle at latest 3 minutes after H3, the 3rd-stage SCAR (Attitude and Roll Control System) enables the composite to execute various manœuvres in accordance with the following standard sequences, selected by the user :

- Sequence A (spin-up) :
  Orientation of the XXn launch vehicle axis in accordance with the direction required by the user, followed by spin-up on this axis to a rate not exceeding 10 rpm and in the direction selected by the user. Execution of these manœuvres takes about 2 minutes.

- Sequence B (3-axis stabilized) :
  Orientation of the launch-vehicle axes according to the directions required by the user. Execution of this manœuvre takes about 1 minute.

In both cases, the desired direction must be defined by reference to the following three axes :

u : radius vector with its origin at the centre of the Earth, and passing through the intended orbit perigee.
v : perpendicular to u in the intended orbit plane, having the same direction as the perigee velocity.
w : perpendicular to u and v, such that u, v, w form a direct trihedron.

The orientation required by the user can be adjusted to take account of the launch time. In this case, the user must supply details of the orientation required at the start and the end of the intended launch window. This orientation will be interpolated between these two values, for a given launch time.

If a type 1194, 937, or 1497 adaptor is used, the launch-vehicle/payload clampband is then released, and the separation actuators impart to the payload a velocity relative to the launcher of at least 0.5 m/s.

If a non-standard adaptor is used (see para. 3.1.1.5.), the relative separation velocity of the payload with respect to the launch vehicle must be at least 0.5 m/s. Paragraphs 3.5.2.3. and 3.5.3. indicate the electrical interface details which must be observed for non-standard adaptors, in particular with respect to command signals from the launch vehicle, and their transmission to the payload-separation devices.

After separation of the payload, the launch vehicle can manœuvre to eliminate any risk of collision.

Figure 3.24 (sequence A for 3 types of typical mission, and 4 spin rates) and figure 3.25 (sequence B for missions described parametrically), give the kinematic conditions of the payload with respect to nominal, immediately after separation, namely when the last actuator of the separation system loses contact with the rear frame of the payload.

3.4. Environment imposed on the payload

3.4.1. Mechanical loads induced by the launch vehicle

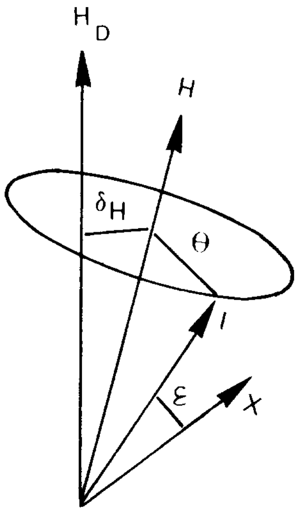
3.4.1.1. General presentation

During flight, the payload is subjected to static and dynamic loads induced by the launch vehicle.

Such excitation may be of aerodynamic origin (wind, gusts, buffeting at transsonic velocity), or due to the stage propulsion systems (longitudinal acceleration, thrust build-up or tail-off transients, structure-propulsion coupling, control limit cycling, etc.).

The loads quoted in the following paragraphs should be considered as extreme in-flight loads, applying to the payload. The corresponding probability of those figures not being exceeded should be taken as 97.7 % (2σ estimate).

In the following paragraphs, the payload base is defined as the top of the payload adaptor used.



H	Actual moment of inertia
H_D	Desired moment of inertia
I	Payload longitudinal inertia main axis
x	Payload longitudinal geometric axis
δ_H	Moment of inertia offset
Θ	Nutation half-cone angle
ε	Dynamic unbalance angle
Ω_X	Spin rate
M_{cu}	Payload mass
X_G	Distance of CG from mounting plane
I_R	Main roll inertia
I_T	Main yaw / pitch inertia
d	Distance of CG from X axis

Mission	Ω_X (rpm)	δ_H max* (degrees)	Θ max* (degrees)	$\delta\Omega_X$ max* (rpm)
M_{cu} = 1700 kg, X_G = 2m, I_R = 800 kg m^2, I_T = 2000 kg m^2, $d \leqslant$ 0.010 m, $\varepsilon \leqslant$ 1 degree	2.5	2.73	3.34	0.17
	5	2.61	3.23	0.15
	7.5	2.78	3.04	0.17
	10	3.69	3.12	0.17
M_{cu} = 1350 kg, X_G = 1.5 m, I_R = 550 kg m^2, I_T = 1100 kg m^2, $d \leqslant$ 0.010 m, $\varepsilon \leqslant$ 1 degree	2.5	2.63	3.10	0.19
	5	1.77	2.62	0.18
	7.5	2.14	2.66	0.19
	10	2.67	2.67	0.17
M_{cu} = 1000 kg, X_G = 0.7 m, I_R = 300 kg m^2, I_T = 300 kg m^2, $d \leqslant$ 0.010 m, $\varepsilon \leqslant$ 1 degree	2.5	2.63	2.72	0.16
	5	1.86	1.70	0.16
	7.5	1.93	1.54	0.19
	10	2.55	1.47	0.18
* 99.73 % maximum value obtained in 1000 cases				

FIG. 3.24 KINEMATIC CONDITIONS AFTER SEPARATION WITH PAYLOAD SPIN-UP (A SEQUENCE)

– Depointing of payload longitudinal geometric axis (parallel to launch vehicle axis XXN) :

mean value : 0.36^{o}

99.73 % value : 1.0^{o}

– Depointing of payload transverse geometric axes :

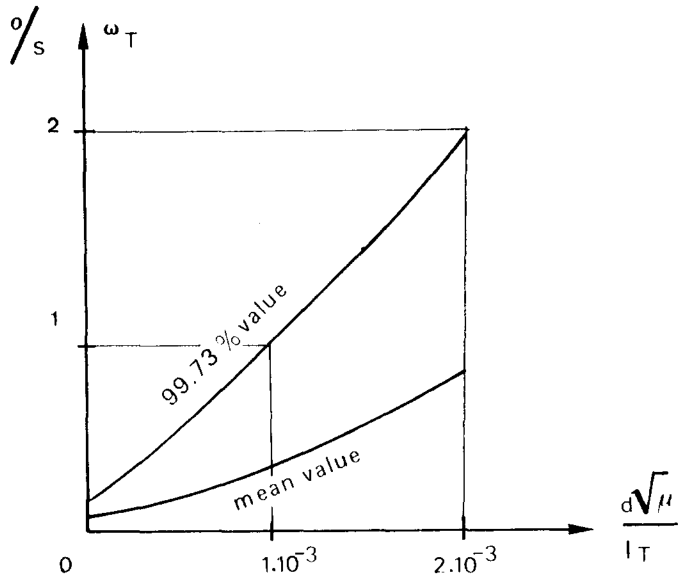
mean value : 0.36^{o}

99.73 % value : 1.0^{o}

– Angular velocity round payload longitudinal geometric axis :

mean value : $0.0^{o}/s$

99.73 % value : $\pm 0.5^{o}/s$



Transverse angular velocity ω_T : ($^{o}/_{s}$)
M_{cu} : payload mass (kg)
I_T : payload lateral inertia (kg.m^2)
d: lateral eccentricity of Cof G (m)

$$\mu = \frac{M_{cu}\, m}{m + M_{cu}} \quad \text{(where m = 1460 kg)}$$

FIG. 3.25 KINETIC CONDITIONS AFTER SEPARATION WITHOUT PAYLOAD SPIN-UP (B SEQUENCE)

3.4.1.2. Static acceleration

The static longitudinal acceleration profile is given in figure 3.26, during the flight of the first two stages. This figure relates to the geostationary mission, but other missions vary only slightly. Maximum longitudinal static acceleration during 3rd-stage flight is 1.8 g, for the geostationary mission.

Lateral acceleration is negligeable, and is therefore ignored.

3.4.1.3. Dynamic environment

3.4.1.3.1. Longitudinal sinusoidal vibrations

The vibration level at the base of the payload is 1.5 g from 10 to 100 Hz. This spectrum takes account of the following :

- pogo-type longitudinal excitation, which can occur during the flight of the 1st stage (11 - 18 Hz) and 2nd stage (28 - 35 Hz)
- transient thrust-decay excitation of the first two stages in the 35 - 100 Hz band, the maximum level occurring between 35 and 50 Hz.

The spectrum is shown in figure 3.27.

3.4.1.3.2. Lateral sinusoidal vibrations

The vibration level at the base of the payload depends on the flight instants considered.

During 1st-stage flight, the larger amplitude vibrations are in the frequency range of the first lateral modes of the vehicle (2 - 15 Hz), which are excited mainly by gusts. Their level is 1.5 g. In the 15 - 100 Hz range, vibrations are caused by transient thrust-decay excitation. Their level is 1 g.

During 2nd-stage flight, the main sources of excitation are transients occurring on engine ignition and cut-off. Response values are higher in the 5 - 18 Hz range, corresponding to the frequencies of the first bending modes. Their level is 1 g. In the 18 - 100 Hz range, their level is 0.6 g.

The sources of excitation during 3rd-stage flight are low, and do not induce dimensioning vibration levels.

The spectrum of lateral sinusoidal vibrations is shown in figure 3.28.

3.4.1.3.3. Random vibrations

Random vibrations are generated by mechanical parts in movement (e.g. turbopumps), combustion phenomena or structural elements excited by the acoustic environment. Such vibrations are transmitted to the payload via the launch-vehicle structures.

The spectrum is shown in figure 3.29, and is applicable to all axes. The rms level is 7.3 g.

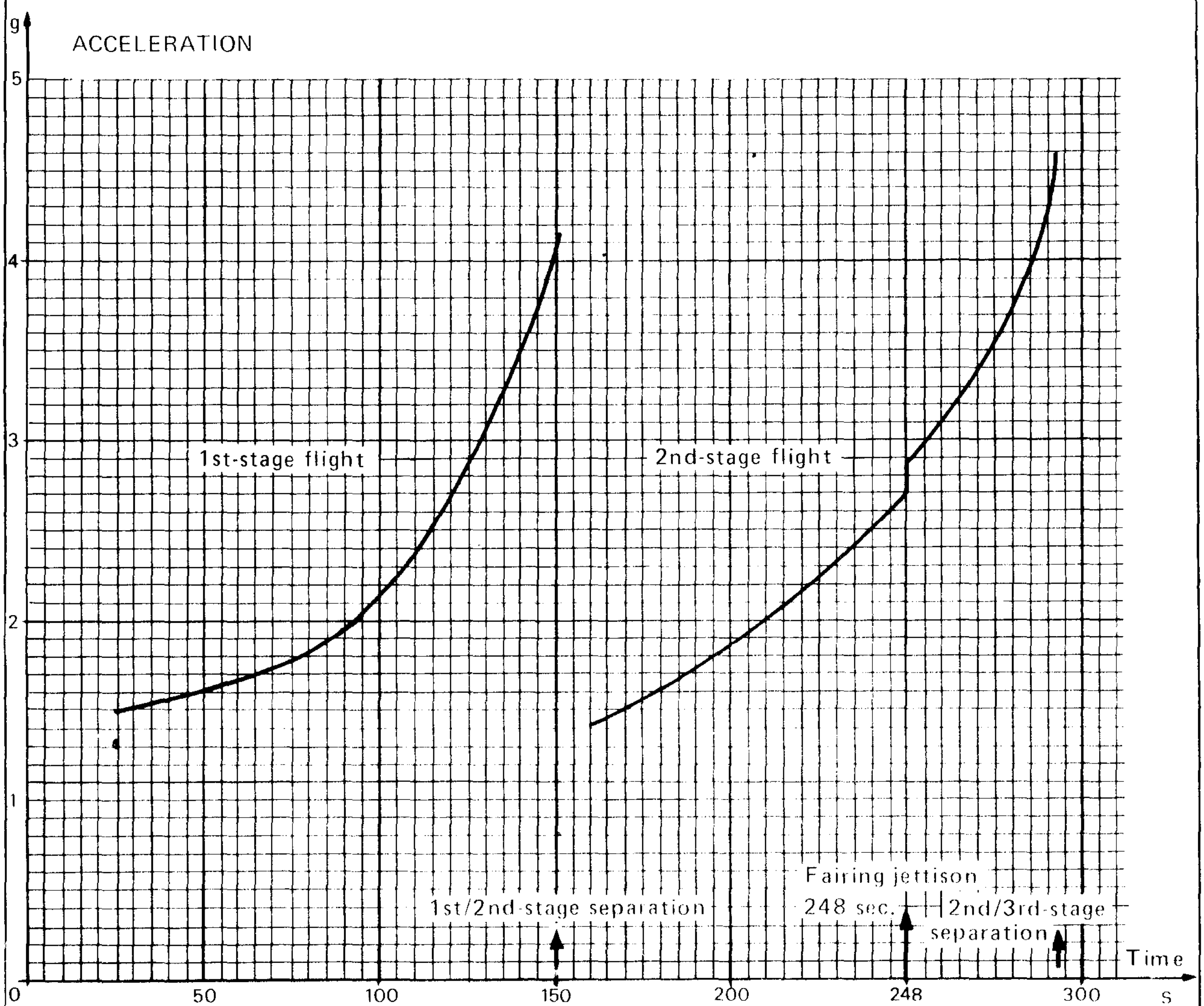


FIG. 3.26 LONGITUDINAL STATIC ACCELERATIONS
GEOSTATIONARY MISSION (FLIGHT LEVELS)

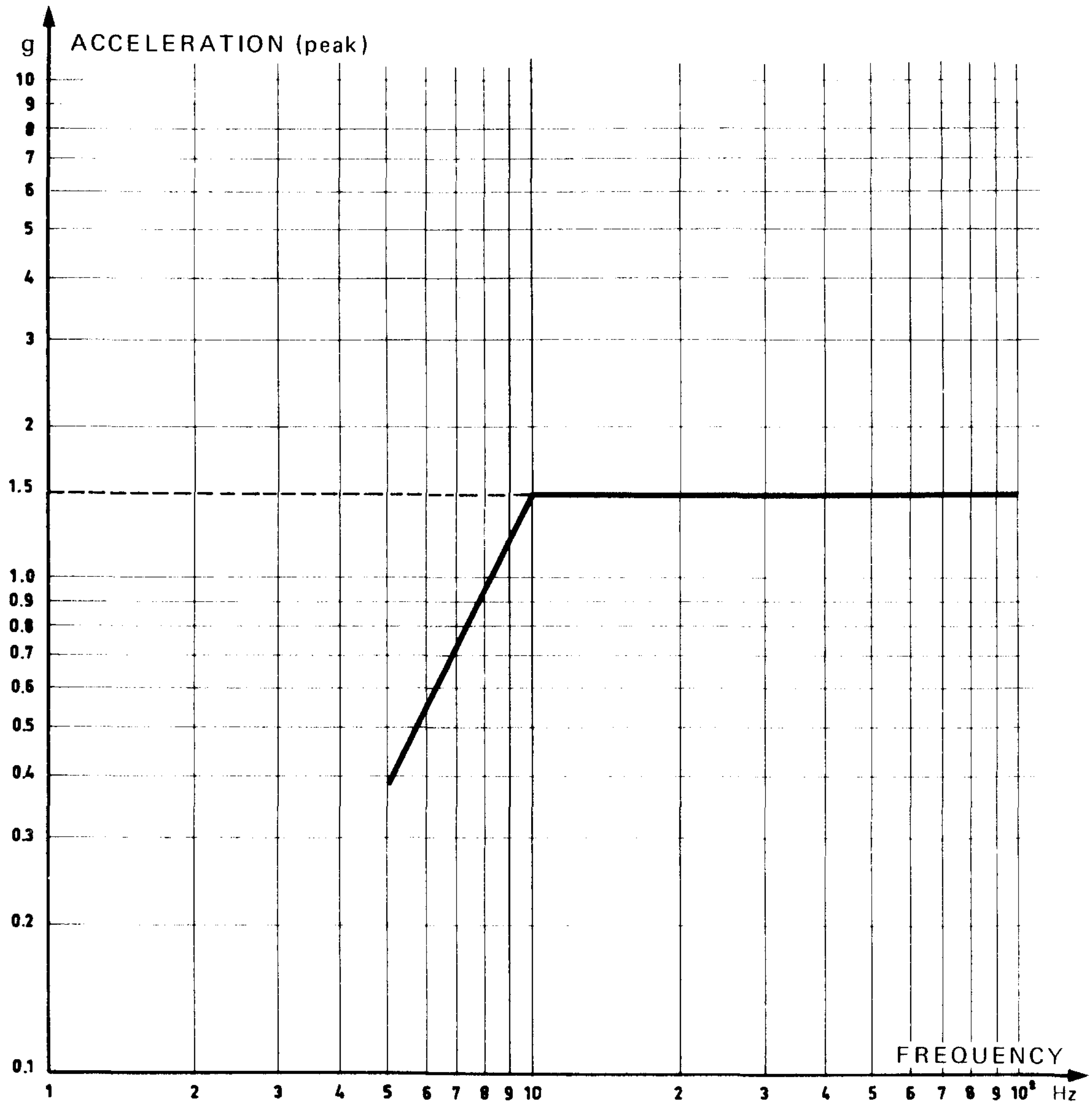


FIG. 3.27 SINUSOIDAL LONGITUDINAL VIBRATIONS (FLIGHT LEVELS)

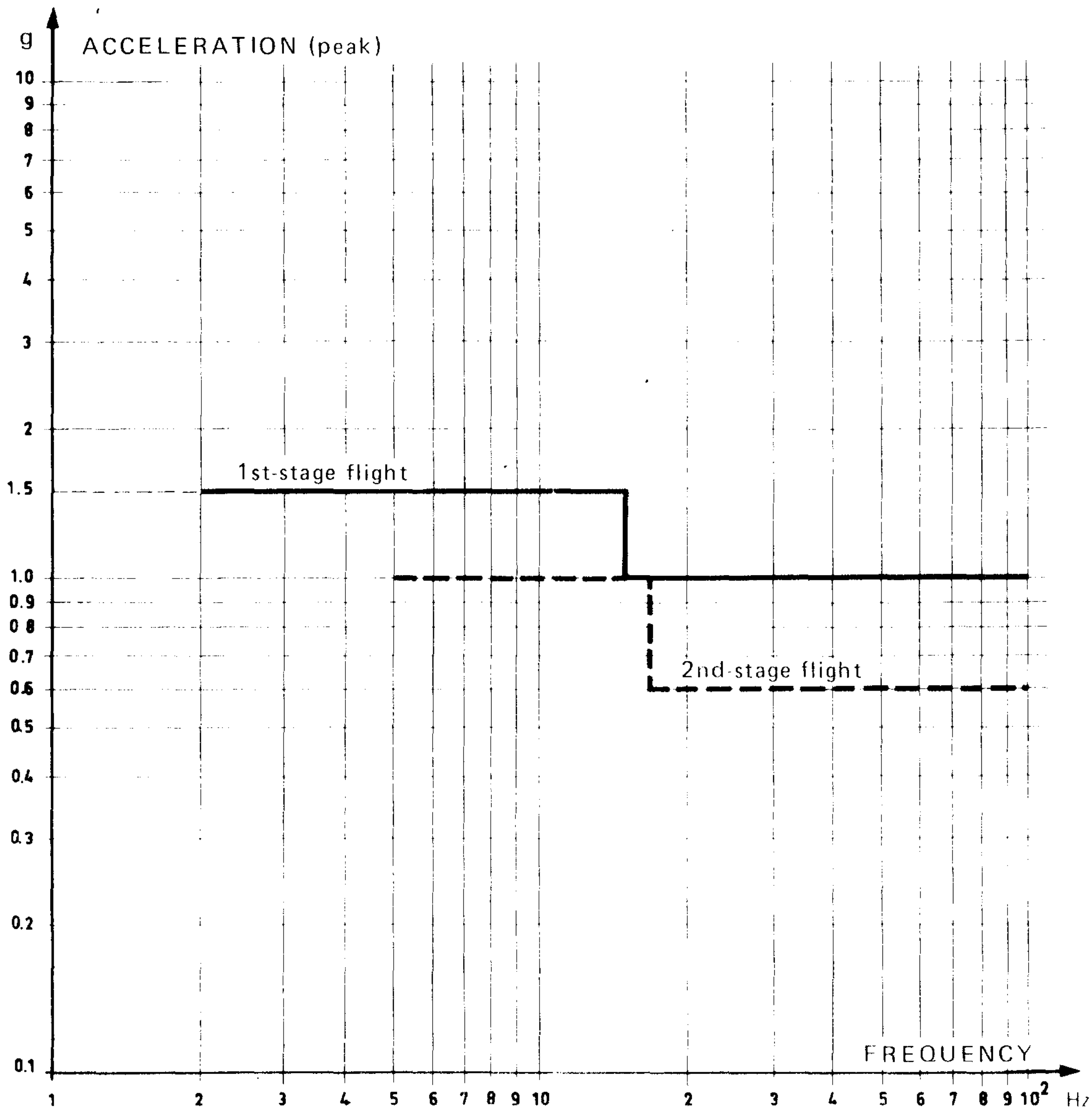


FIG. 3.28 SINUSOIDAL LATERAL VIBRATIONS (FLIGHT LEVELS)

3.4.1.3.4. *Acoustic vibrations*

Acoustic vibration is generated by engine noise, buffeting and boundary-layer noise. It is maximum at lift-off and in the vicinity of the transsonic (about 66 s). Outside these periods it is substantially lower.

The maximum values encountered on the payload are shown in figure 3.30, whether the fairing is equipped with acoustic protection or not.

3.4.1.3.5. *Shocks*

The payload is subjected to shocks principally during separation of the fairing, and on actual payload separation.

Shock excitation occurs above 400 Hz.

3.4.1.4. Payload design and dimensioning

3.4.1.4.1. *Frequency requirements*

To avoid dynamic coupling between the launch vehicle and the payload, steps must be taken to ensure that :

- the fundamental bending-mode frequency of the payload, assumed to be hard-mounted, is greater than 10 Hz.
- the frequencies of the main longitudinal modes exceed 40 Hz.

It should also be noted that transient thrust-decay excitation occurring in the 35 - 50 Hz band can affect the secondary structures and flexible elements (e.g. antennae and solar panels), which should therefore be designed so that their natural frequencies are outside this band.

3.4.1.4.2. *Primary-structure dimensioning loads*

During flight, the various static and dynamic loads are superimposed. The design and dimensioning of the payload primary structure must therefore allow for the most severe load combination that may be encountered at a given instant of flight. From the point of view of quasi-static loads, the most critical instants of flight from the dimensioning point of view are as follows :

- maximum dynamic pressure,
- 2nd-stage burn-out.

The following table combines low-frequency dynamic and static acceleration for these events. The longitudinal and lateral accelerations indicated in the table act simultaneously at the centre of gravity of a typical Ariane 1700-kg payload that meets the above-mentioned frequency requirements and whose centre of gravity is located 2 m above the payload mounting plane.

The following loads should be used for the preliminary definition of the payload.

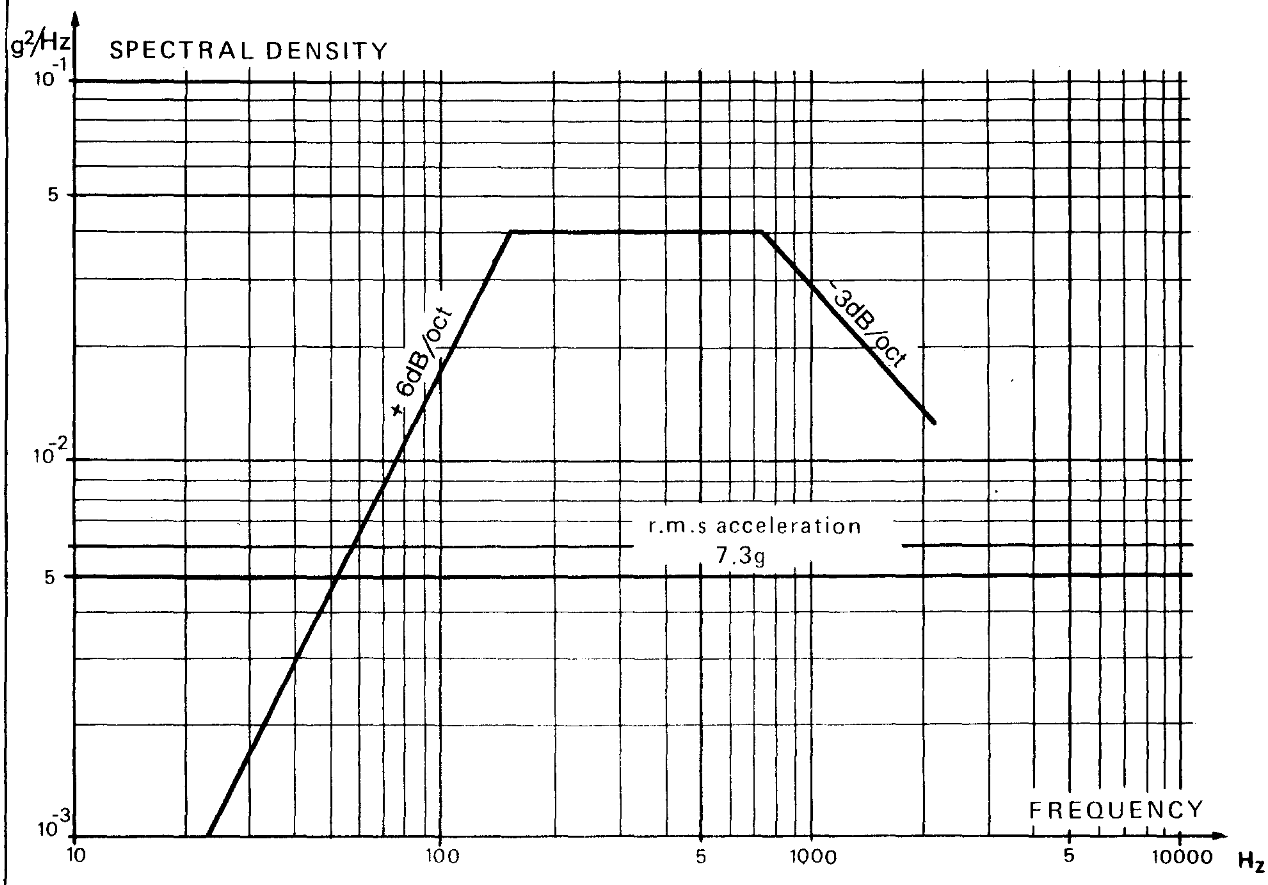
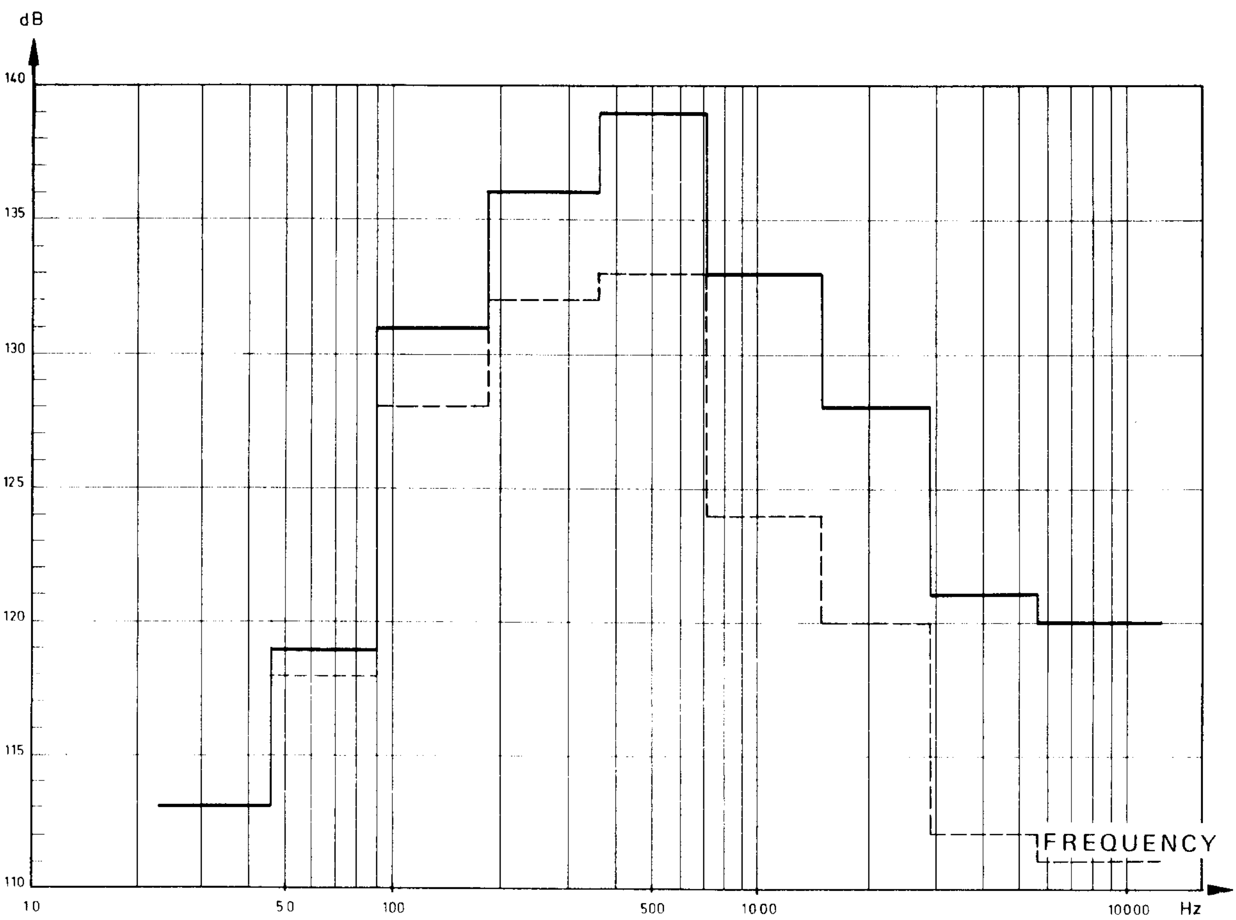


FIG. 3.29 RANDOM VIBRATIONS (FLIGHT LEVELS)



without acoustic blanket INTEGRATED LEVEL 142 dB

with acoustic blanket INTEGRATED LEVEL 137 dB

FIG. 3.30 NOISE SPECTRUM UNDER FAIRING

Flight limit loads

Acceleration (g) / Flight event	Longitudinal axis			Lateral axis
	Static	Dynamic	Total	
Maximum dynamic pressure	− 1.9	± 1.6	− 3.5	± 2
2nd stage burn-out	− 4.7 + 0	± 3.2	− 7.9 + 3.2	± 1
The minus sign for longitudinal-axis values indicates compression. Lateral loads may act in any direction. The above loads are only valid if type 1194, 937 or 1497 adaptor are used (see para. 3.1.1.1), or alternatively adaptors meeting the requirements of para. 3.1.1.5.				

These accelerations are applied to the greater part of the payload, but certain areas are subjected to higher accelerations. The distribution of acceleration within the payload depends on the configuration of the latter, and the Ariane authority must be consulted on this subject.

It is advisable to carry out a preliminary coupled analysis at this stage, in particular for payloads the characteristics of which would not meet the requirements indicated in para. 3.4.1.4.1.

Dimensioning must take account of safety factors, which will be defined by the payload authority (the Ariane authority calls for a minimum value of 1.25 at rupture).

3.4.1.4.3. *Dimensioning of secondary structures and flexible elements*

The secondary structures and flexible elements (e.g. solar panels, antennae and propellant tanks) must be designed to withstand the dynamic environment induced at the base of the payload (as described in para. 3.4.1.3.), taking account of amplification resulting from the design and arrangement of the payload.

A safety factor of at least 1.5 will be assumed for the dimensioning of these structures. The strength of these elements and their attachment to the primary structure will be verified by dynamic-environment tests.

3.4.1.4.4. *Coupled analysis*

In all cases, coupled analysis carried out in connection with mission analysis (see chapter 6) will have to confirm the chosen dimensioning.

3.4.1.5. Payload qualification and acceptance

The payload authority must prove that the payload meets the dimensioning requirements detailed in para. 3.4.1., by means of a calculation file and qualification tests (see chapter 6).

The elements needed for the test plan are described in the following paragraphs. The test plan will be discussed with the Ariane authority.

The acceptance levels are representative of the flight environment described in para. 3.4.1.3.

3.4.1.5.1. Static qualification tests

On the basis of the study of combined loads, as described in para. 3.4.1.4.2., the user determines the dimensioning load cases to which the payload structure will be subjected. The static tests will be carried out up to flight limit loads, multiplied by the structure-rupture safety factor.

3.4.1.5.2. Sinusoidal vibration tests

It is recommended that a factor of 1.5 be used for qualification tests.

The qualification and acceptance levels applied at the base of the payload are set out in the following table and in figure 3.31.

	Frequency range (Hz)	Qualification level (0-peak)	Acceptance level (0-peak)
Longitudinal	5 - 10 10 - 100	5.7 mm 2.25 g	3.8 mm 1.5 g
Latéral	5 - 7 7 - 15 15 - 100	11.6 mm 2.25 g 1.5 g	7.7 mm 1.5 g 1.0 g

If a payload does not meet the lateral-vibration requirements at frequencies above 10 Hz, the figures given above for lateral vibration at low frequencies will have to be redetermined.

It is advisable for each test to be preceded by a low-level vibration test, in order to detect payload resonant frequencies and check possible notching procedures.

The sweep rate is 2 octaves/minute for qualification, and 4 octaves/minute for acceptance testing.

3.4.1.5.3 Notching

Notching procedures are generally accepted during sinusoidal vibration testing. The following principles shall apply to their execution :

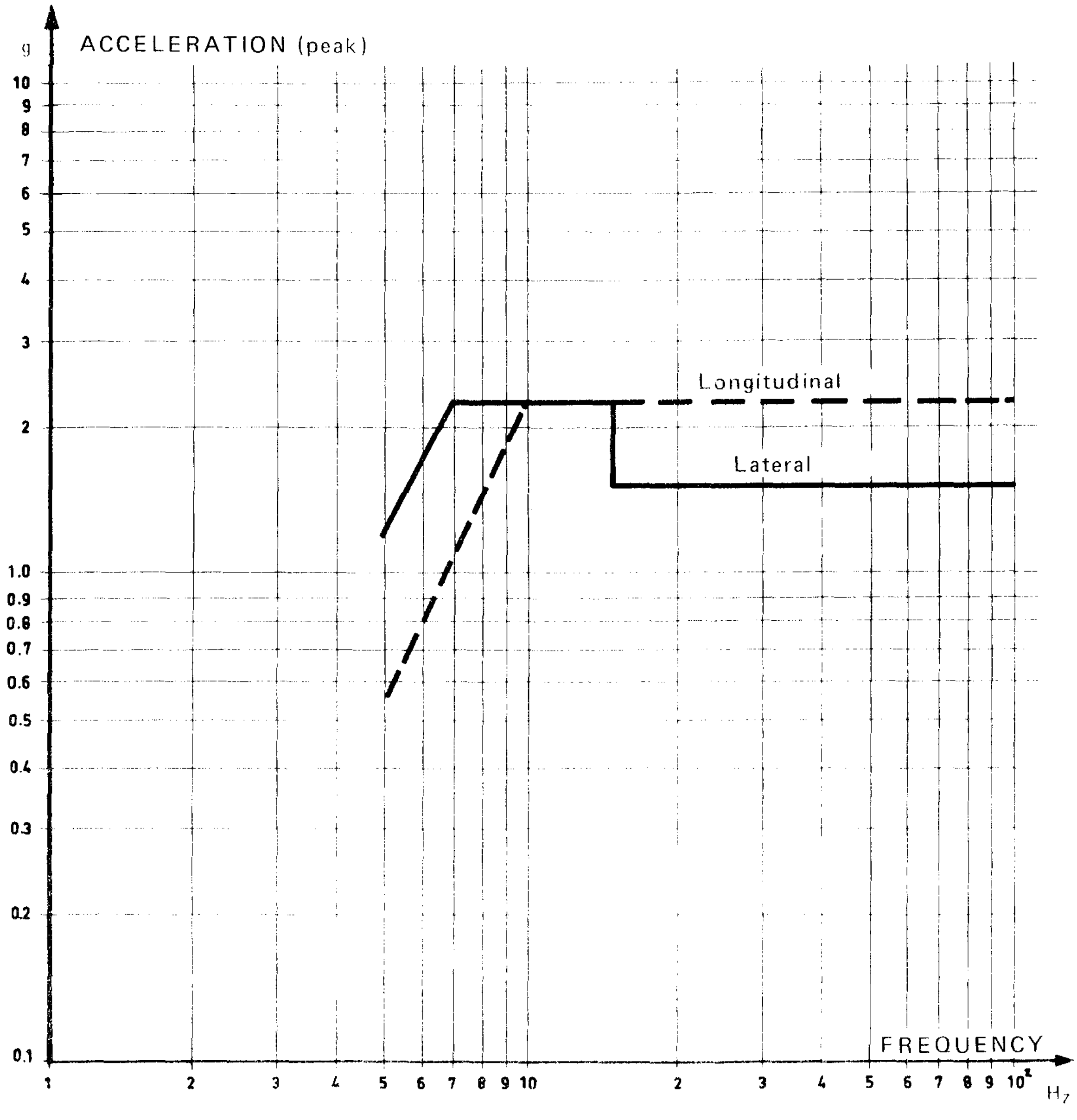


FIG. 3.31 – SINUSOIDAL ACCELERATIONS
QUALIFICATION LEVELS

a) The excitation level during the sinusoidal vibration test can be reduced, so as to remain within the dimensioning limits for the primary structure.

b) In the light of the requirements of para. 3.4.1.4.1., notching is not authorized for longitudinal sinusoidal tests in the 11 - 35 Hz band.

c) During lateral sinusoidal testing, the maximum bending moment at the base of the payload can be limited to the maximum bending moment determined by coupled analysis.

d) Notching procedures authorizing a reduction in vibration levels at the resonant frequencies of secondary structures or flexible elements (antennas, solar panels, etc.) can only be allowed in the light of the results of coupled analysis. In this case, modelling of the payload must be representative of the secondary structure or flexible elements concerned, and be validated by results of low-level tests.

These notching procedures must be discussed with the Ariane authority, and indicated in the payload environmental test plan (see para. 6.1.4.).

3.4.1.5.4. *Random vibration tests*

The random-vibration test levels are as follows :

	Frequency range (Hz)	Density (g^2/Hz)	rms value (g)
Qualification	30 - 150 150 - 700 700 - 2000	+ 6 dB/Oct 0.09 – 3 dB/Oct	11
Acceptance	30 - 150 150 - 700 700 - 2000	+ 6 dB/Oct 0.04 – 3 dB/Oct	7.3

The levels are identical for longitudinal and lateral vibration. Test time is 3 minutes per axis for qualification, and 2 minutes per axis for acceptance testing.

3.4.1.5.5. *Acoustic vibration tests*

The time for the acceptance tests is 1 minute. The levels indicated in the following table allow for test-equipment accuracy.

	Level dB (ref. 2.10^{-5} Pascal)			
Octave band ((Centre frequency) (Hz)	Without acoustic protection	Test tolerance	With acoustic protection	Test tolerance
31.5	113	− 3/ + 5	113	− 3/ + 5
63	119	− 2/ + 4	118	− 2/ + 4
125	131	− 2/ + 4	128	− 2/ + 4
250	136	− 2/ + 4	132	− 2/ + 4
500	139	− 2/ + 4	133	− 2/ + 4
1000	133	− 2/ + 4	124	− 2/ + 4
2000	128	− 2/ + 4	120	− 2/ + 4
4000	121	− 5/ + 5	112	− 5/ + 5
8000	120	− 5/ + 5	111	− 5/ + 5
Overall level	142	− 1/ + 3	137	− 1/ + 3

The qualification test duration is 2 minutes, levels being 4 dB greater than the acceptance levels given above.

3.4.1.5.6. *Shock tests*

The shock test involves igniting the pyrotechnic separation system, in the presence of the payload adaptor, twice for qualification and once only for acceptance testing.

3.4.2. Thermal and climatic environment

3.4.2.1. Environment in payload facilities

(See para. 4.6 and fig. 4.28).

3.4.2.2. Ground environment within the fairing

After the fairing has been fitted, the payload compartment is air-conditioned (see fig. 3.32) throughout the launch operations, both outside and inside the tower. This ventilation, which is supplied via the fairing umbilical, is continuous until lift-off. It characteristics are as follows :

- temperature of air injected under the fairing .... adjustable between 15° and 20° C
- relative humidity .... < 15 %
- filtration .... class 100 000 (US Fed Std 209b)
- air flow .... 3 000 kg/hour
- air circulation velocity .... < 2 m/s

FIG. 3.32 COOLING UNDER FAIRING
(Will be published in a subsequent issue)

The temperature of the air injected under the fairing can be adjusted at the user's request, between 15° and 20° C, once and once only during the waiting time.

The temperature of the fairing inner face never exceeds 35° C, irrespective of waiting conditions (inside or outside the tower).

Typical temperatures for the VEB equipment and the diaphragm separating the 3rd stage from the VEB are given in figure 3.33.

The noise level generated by the ventilation system does not exceed 80 dB as a general rule. Transient phenomena during start-up or shutdown may generate higher levels. Any such transients only last a few seconds.

Local ventilation can be provided at the user's request.

3.4.2.3. In-flight environment within the fairing

3.4.2.3.1. Temperature and flux variations

The flux density radiated by the fairing and the VEB does not exceed 500 W/m² at any point. Figure 3.34 shows how the structure temperature varies at different points of the fairing, with and without acoustic protection, and figure 3.35 shows the corresponding variation of radiated flux density, the emissivity (ε) of the internal surfaces being :

nose :	= 0.1
forward cone and cylinder :	= 0.2
rear cone :	= 0.7
acoustic protection :	= 0.8

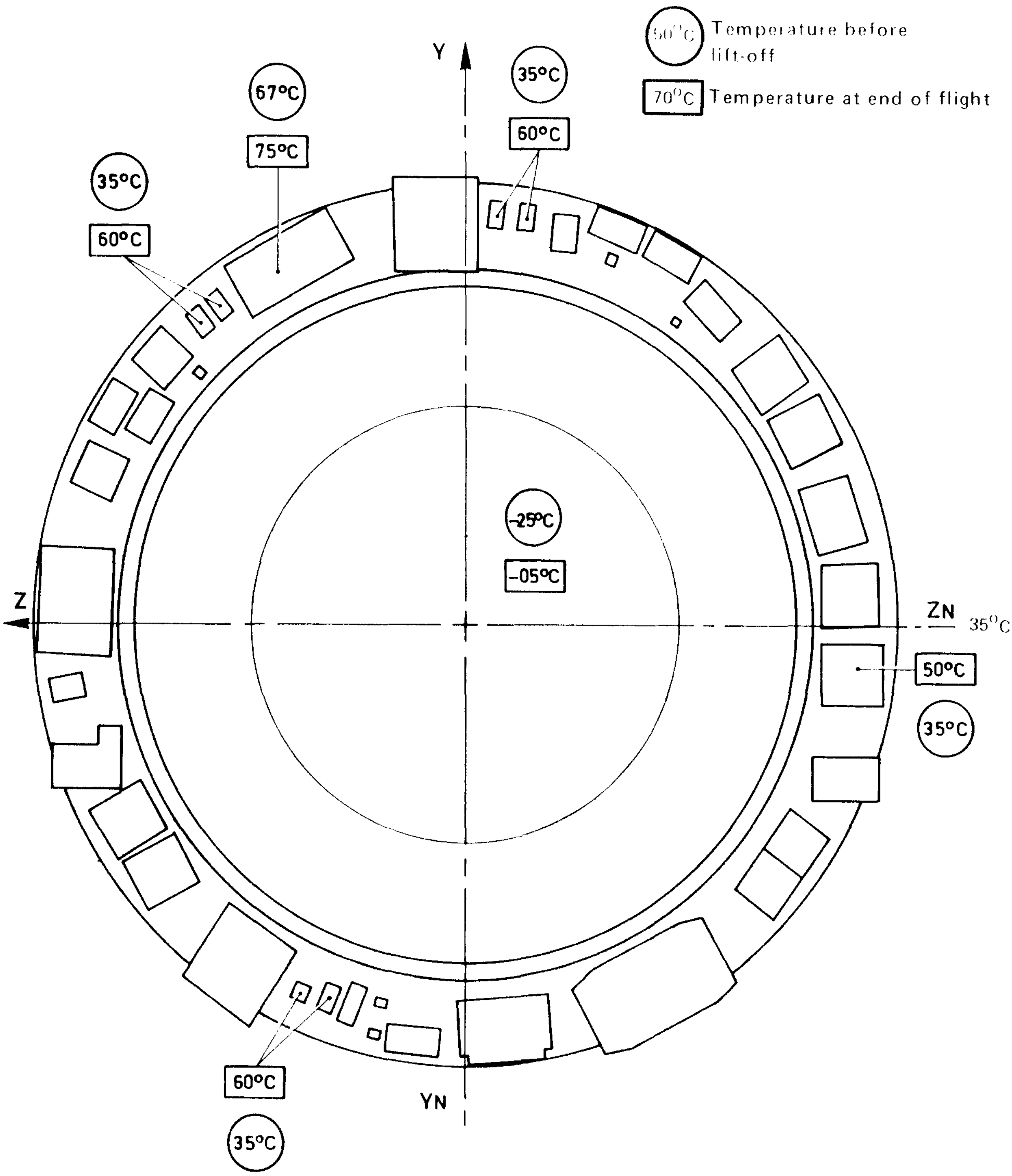
These values are calculated for a trajectory corresponding to a geostationary mission, assuming a dispersion for all propulsion characteristics of two standard deviations.

3.4.2.3.2. Variation of static pressure within the fairing

Figure 3.36 shows the variation of static pressure within the fairing, as an envelope curve for the various typical missions described in chapter 2.

3.4.2.4. Aerothermal flux at fairing jettison.

For all missions, the nominal instant of jettisoning the fairing is determined so that aerothermal flux, calculated as a free molecular flow acting on a plane surface perpendicular to the direction of velocity (atmosphere US 66, latitude 15° North), is less than 1135 W/m², with a probability of 99.87 %.



- Equipment temperatures are indicated only if they exceed 45° at end of flight or 35° before lift-off
- Structure temperature is lower than 35°C
- Structure and VEB equipment emissivity $\varepsilon = 0.9$
  Membrane emissivity $\varepsilon = 0.7$

FIG. 3.33 – VEB TEMPERATURE MAP

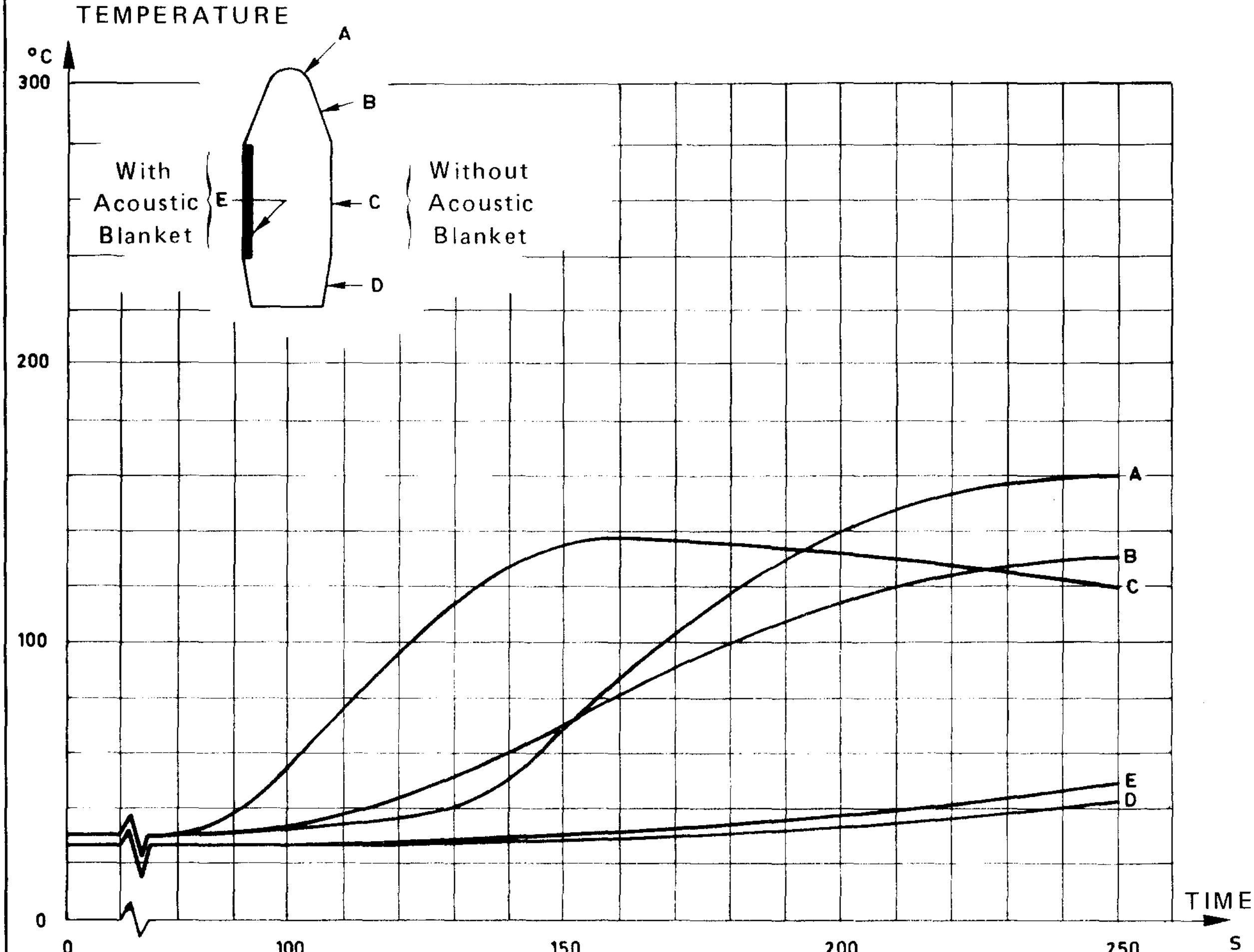


FIG. 3.34 INTERNAL TEMPERATURES OF FAIRING STRUCTURE
WITH AND WITHOUT ACOUSTIC BLANKET

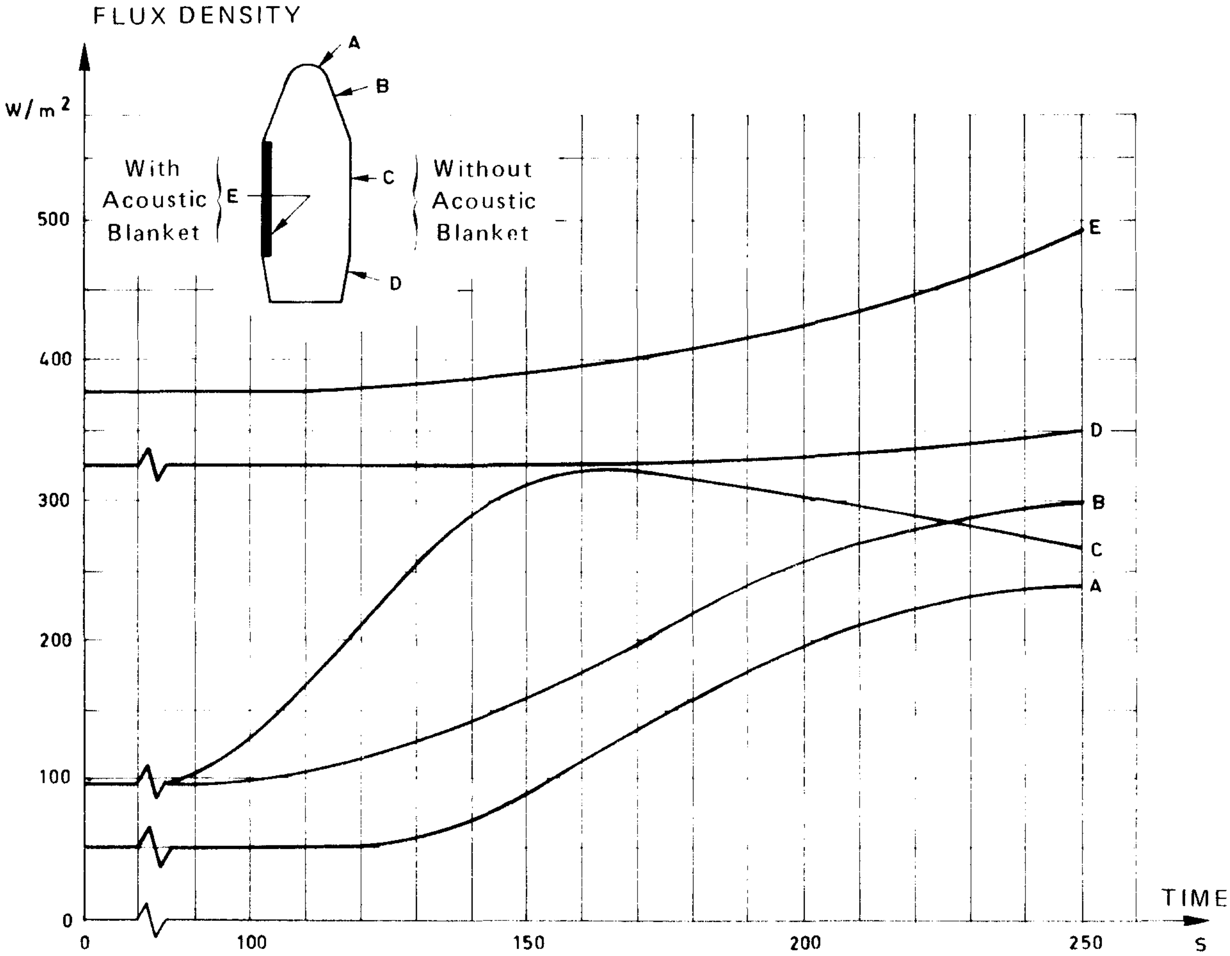


FIG. 3.35 FLUX EMITTED BY FAIRING STRUCTURE WITH AND WITHOUT ACOUSTIC BLANKET

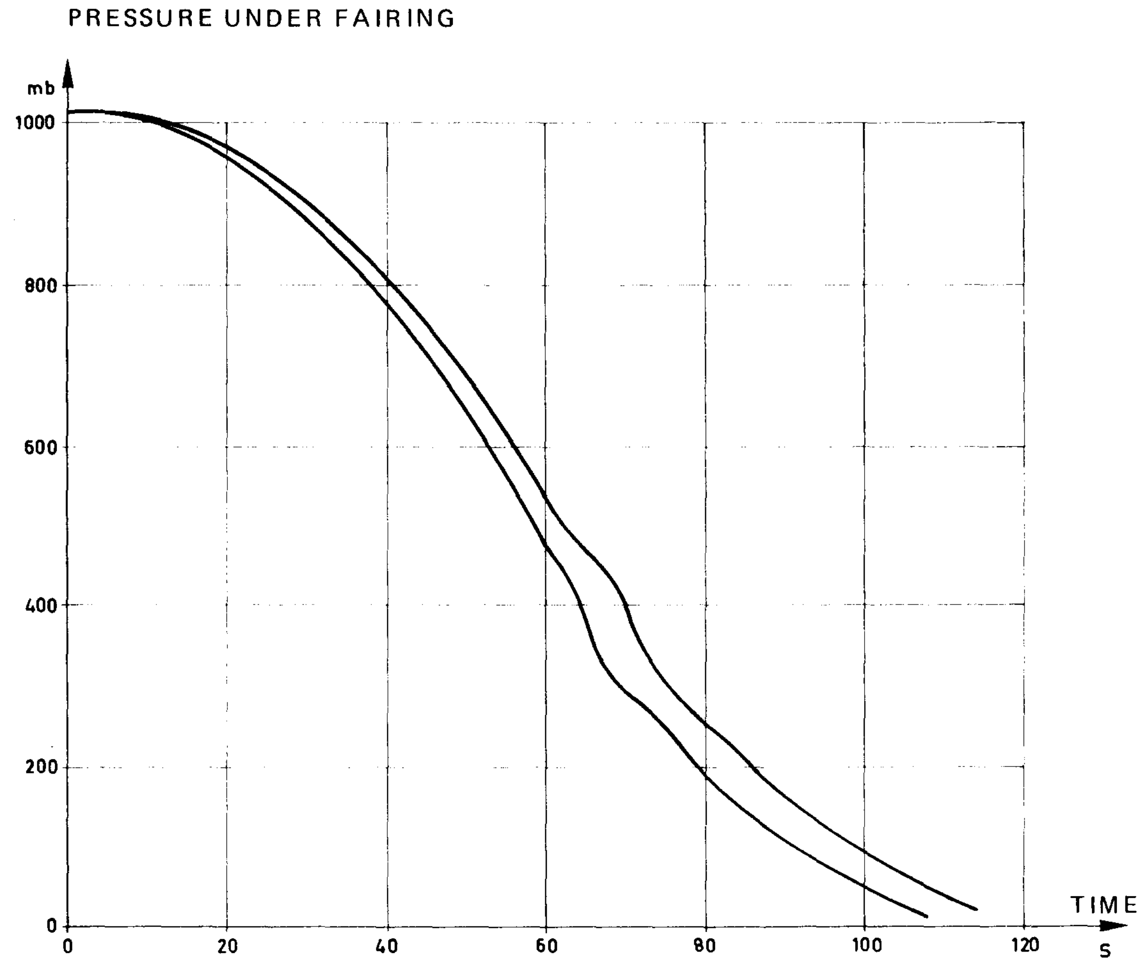


FIG. 3.36 PRESSURE UNDER FAIRING

The performance values given in chapter 2 are calculated on the basis of the jettison conditions defined above. An aerothermal flux value other than 1135 W/m² may be obtained by varying the instant of jettisoning the fairing.

Solar-radiation flux, albedo and terrestrial infrared must be added to this aerothermal flux. In calculating the incident flux on the payload, account must be taken of the altitude of the launch vehicle, its orientation (see figs. 2.9, 2.11, 2.15 and 2.17), the position of the sun with respect to the launch vehicle, and the orientation of the payload surfaces considered.

3.4.3. Contamination

The materials used in the upper part of the launch vehicle close to the payload (e.g. VEB and fairing) do not generate an organic deposit on the payload of more than 2 mg/m², measured in accordance with procedure PSS-15/QRM-05 T (" Detection of organic contamination of surfaces by infrared spectroscopy techniques ").

The acoustic insulation of the fairing is such that outgassed mass is less than 1 % of total mass, and condensable volatile materials are less than 0.1 % of total mass, measured in accordance with procedure PSS-09/QRM-02 T, Issue 2 (" A screening test method employing a thermal vacuum for the selection of materials to be used in space ").

The integrated particle flux emitted by the pyrotechnic system and retro-rockets of the 2nd stage is less than 1 g/m² at payload level. These systems do not generate organic deposits exceeding 2 mg/m² on the payload.

The fairing-separation pyrotechnic system is leak-proof and does not cause any contamination.

3.5. Electrical and radio interface

3.5.1. Earth-potential continuity

Close to the separation plane, the payload must have an Earth reference point on which a test socket can be mounted. The resistance between any metal element of the payload (equipment housings, structure, etc.) and the reference point must be less than 10 mΩ for a current of 10 mA.

No payload surface in contact with the adaptor must undergo any treatment or protective process creating a resistance greater than 10 mΩ for a current of 10 mA, between the payload reference point and that of the adaptor.

3.5.2. Payload umbilical

3.5.2.1. Standard umbilical for single launch

3.5.2.1.1. Adaptor type 1194

The standard payload umbilical link is via 2 connectors, P1 and P2, located in diametrically opposite positions in the payload/adaptor separation plane.The mechanical interface for these umbilical connectors is defined in figures 3.3c, 3.5 and 3.6c, figure 3.7 illustrating the principle of the umbilical link.

At the launch-vehicle end, the link has two Deutsch 37-pin male connectors (DBAS 79-37 O-PN) for the standard version. These connectors are wired to the VEB external umbilical plug :

- via cables specific to the adaptor, terminating in interface connectors,
- via standard VEB cables linking the interface connectors to the external umbilical plug.

These interface connectors are used to separate the wires providing the umbilical links to the external umbilical plug, the telemetry acquisition unit (see para. 3.5.4.), and the VEB command unit (see para. 3.5.2.3.).

The Ariane authority will supply the user systematically with the female parts (Deutsch DBAS 70-37 OSN) of umbilical connectors P1 and P2 to be mounted on the payload flight model. Maximum insertion or extraction load is 230 N per connector.

Figure 3.38 indicates the availability of umbilical connections between the VEB and the VEB mast junction box.

3.5.2.1.2. Adaptor type 937

For a single launch of a payload using a type 937 adaptor, the latter incorporates an umbilical link with two connectors, P1 and P2, located in diametrically opposed positions in the separation plane. The mechanical interface for these umbilical connectors, outside the payload lower frame, is defined in figures 3.7, 3.9, 3.11 and 3.12c.

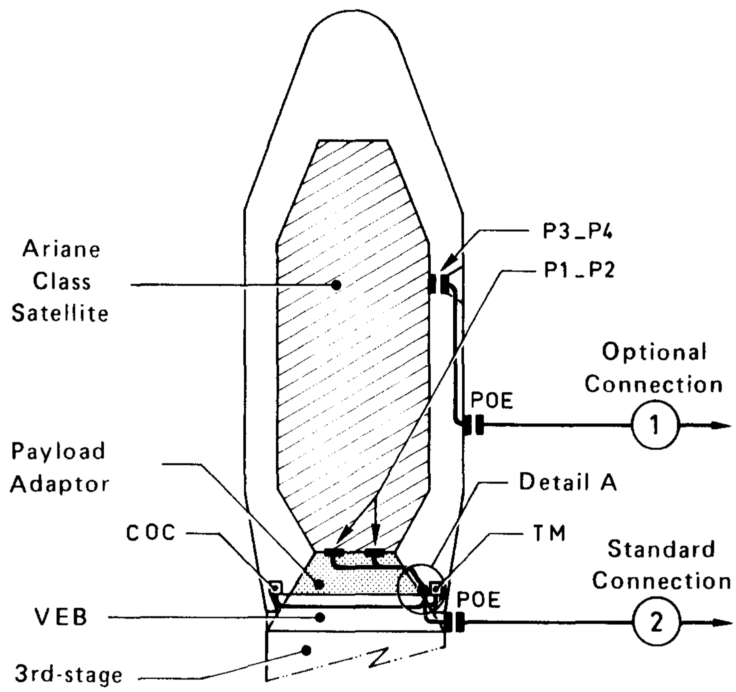
3.5.2.1.3. Adaptor type 1494

TBD

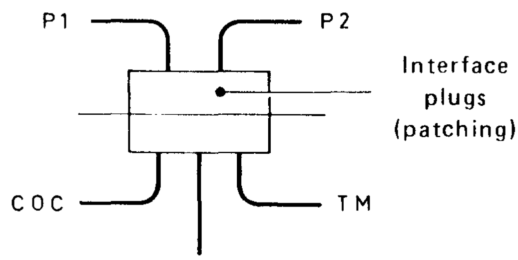
3.5.2.2. Optional umbilical for single launch

The user can be provided with an umbilical link passing through the fairing. At the fairing end, this link has one (or two) 27-pin male connectors, P3 (and P4), types Deutsch DBAS 78-27-0-PN and DBAS 78-27-0-PW respectively. Figure 3.39 shows the area authorized for the payload connectors, and the mechanical interface. These connectors are linked to the fairing external umbilical plug via cables made up to suit user requirements (types and numbers of conductors).

SINGLE LAUNCH



_ DETAIL A _



Note: For connections to the umbilical mast, see Fig. 7.21

FIG. 3.37 — PRINCIPLE OF UMBILICAL CONNECTIONS

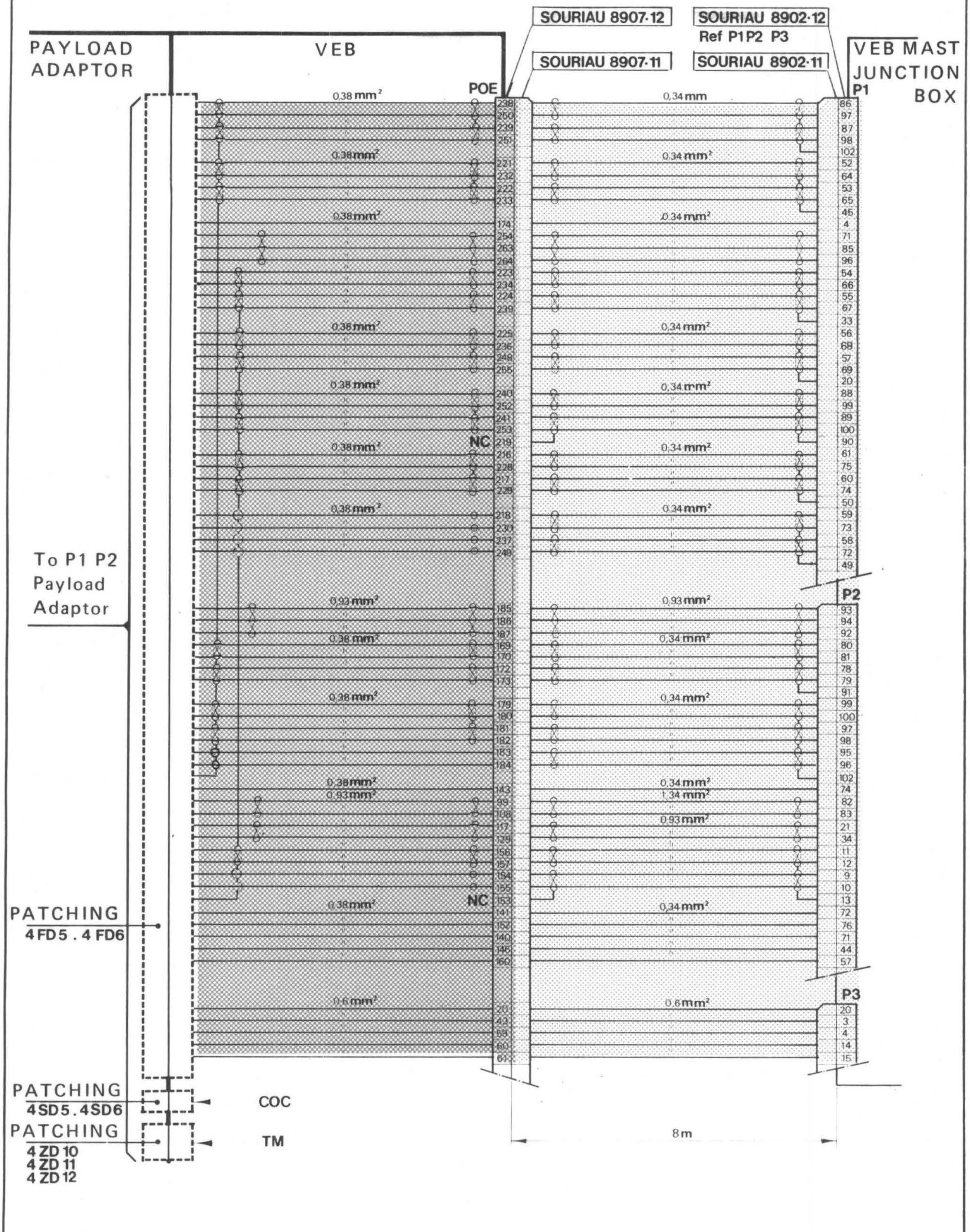


FIG. 3.38 UMBILICAL CONNECTIONS VEB/VEB MAST JUNCTION BOX

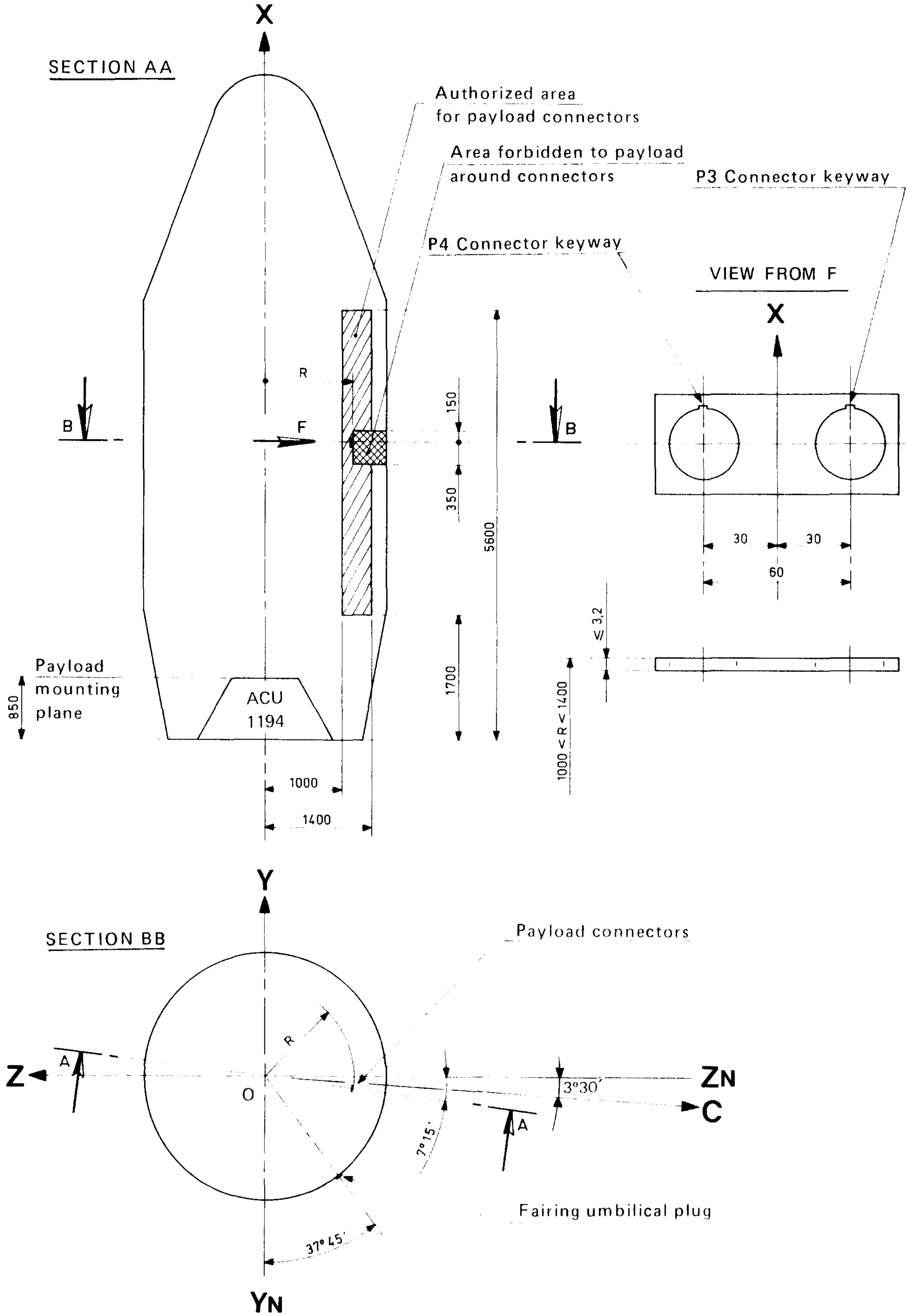


FIG. 3.39 – POSITION OF FAIRING UMBILICAL PLUGS (OPTIONAL)

Connectors P3 and P4 are ejected in flight, at fairing jettison, parallel to the 0C axis (see fig. 3.39, with a tolerance of $\pm$ 5°. Maximum pull on extraction is 180 N per plug.

The Ariane authority can supply the user with the female parts (Deutsch DBAS 70-27-0-SN and 70-27-0-SW) of umbilical connectors P3 and P4, for mounting on the payload flight model.

Figure 3.40 shows the availability of connections between the fairing and the payload mast junction box.

3.5.2.3. Separation commanded by payload.

A redundant pyrotechnic command can be generated by the VEB command unit, on an order from the on-board computer, to control the separation system (maximum 3 initiators : 1A, 1W, no-fire), supplied by the payload. In this case, the interface must be defined case-by-case with the Ariane authority.

3.5.2.4. Standard umbilicals for Sylda dual launch

(see chapter 7)

3.5.3. Pyrotechnic and electrical commands generated by VEB

Characteristics of pyrotechnic commands (for initiator 1A, 1W, no-fire) :

- battery voltage : 14.4 $\pm$ 2 V
- current per initiator : 3 to 3.5 A
- command length : 300 ms

Characteristics of electrical commands :

- 3 redundant electrical commands, with their associated ground commands, can be assigned to the payload :
  - voltage : 28 $\pm$ 4 V
  - current : $\leqslant$ 0.5 A

3.5.4. Payload measurements transmitted by launch vehicle

The VEB telemetry system sends status signals concerning satellite separation, provided by microswitches on the launcher.

In the case of the type 1194 adaptor, two pins of one of the two connectors must be strapped at the payload end.

In the case of the type 937 adaptor, microswitches located inside the springs that distance the satellites enable telemetering of the separation.

In the case of a single-payload launch for which the optional umbilical link via the fairing has been requested, no provision is made for telemetering payload separation.

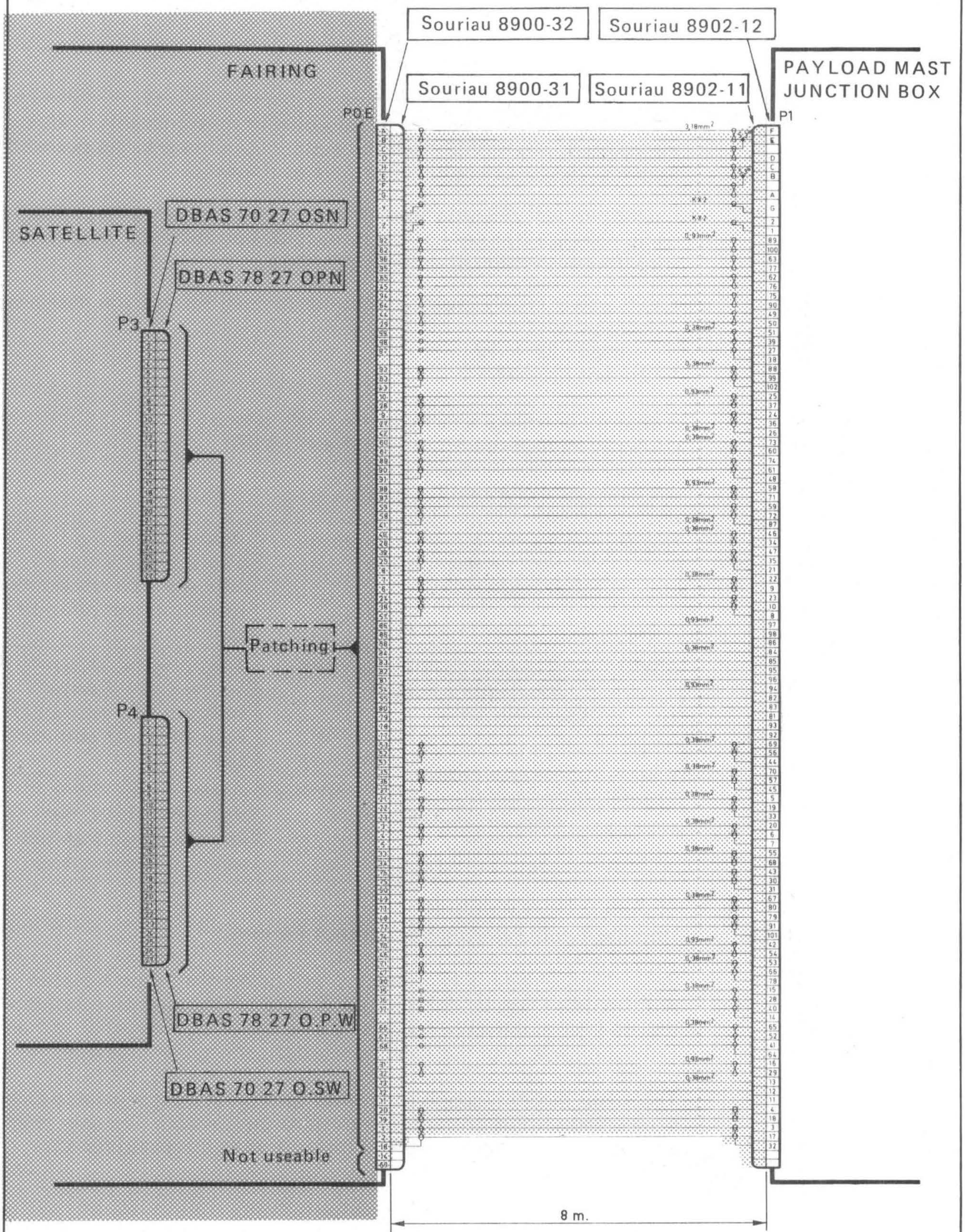


FIG. 3.40 UMBILICAL CONNECTIONS FAIRING / PAYLOAD MAST JUNCTION BOX

The user can call for in-flight transmission of measurements made on the payload, by the VEB telemetry system using transducers powered from the VEB. Interface conditions are as follows :

- in the payload/VEB separation plane, a number of wires in the payload/VEB umbilical links can be used for supplying power to the sensors and for the transmission of measurement data.
- VEB power-supply characteristics for payload sensors (on interface connectors) :
  - voltage : 28 $\pm$ 4V
  - current : $\leqslant$ 0.5A
- capacity of measurement plan for single launch :
  9 two-wire measurements (HNBD and BNBD)
  6 one-wire measurements (HNMD)
  7 time-signal channels ($\leqslant$ 0.5V)

HNBD :	*High Level Bipolar Unbalanced ("Haut niveau bipolaire dissymétrique")*	*Analog signal, 0-5 V, bipolar input. Source balanced with respect to acquisition-unit zero voltage*
BNBD :	*Low Level Bipolar Unbalanced ("Bas niveau bipolaire dissymétrique")*	*Analog signal, 0-20 mV, bipolar input. Source balanced with respect to acquisition-unit zero voltage*
HNMD	*High Level Monopolar Unbalanced ("Haut niveau monopolaire dissymétrique")*	*Analog signal, 0-5 V, monopolar input, with low connected to acquisition-unit zero voltage*

3.5.5. Umbilical cable for testing without fairing.

When the payload has been erected, and before the fairing is placed in position, link 1 (see fig. 3.37) is established from the payload mast junction box, using the umbilical cable connected to a patch panel, and an extension cable from this panel to payload connectors P3 and P4.

The Ariane authority supplies the umbilical cable and patch panel, which is mounted on PF8 of the tower close to the payload. The user must then supply the cable connecting his payload to this patch panel.

3.5.6. Payload mast junction box - Launch Centre Connections

The connection from the payload mast junction box to the Launch Centre comprises the following parts :

- mast cable : connecting the mast junction box to the patchboard at the Forward Test Post (FTP) near the foot of the mast. These cables are approximately 150 m long.
- ground cables connecting the FTP patchboard to the patchboard in the Launch Centre. These cables are approximately 250 m long.
- internal cables in the Launch Centre

Figure 3.41 shows the availability of connections between the payload mast junction box and the Launch Centre.

Patching of connections between the payload check-out racks in the Launch Centre and a payload mounted on the launch vehicle is via :

- payload patchboard in the Launch Centre,
- patchboard in the forward test post,
- payload mast junction box,
- VEB mast junction box for single or lower satellite.

3.5.7. Connections with the launch-vehicle checkout system

Connections can be established from the payload patchboard to the checkout system, providing the following possibilities during countdown :

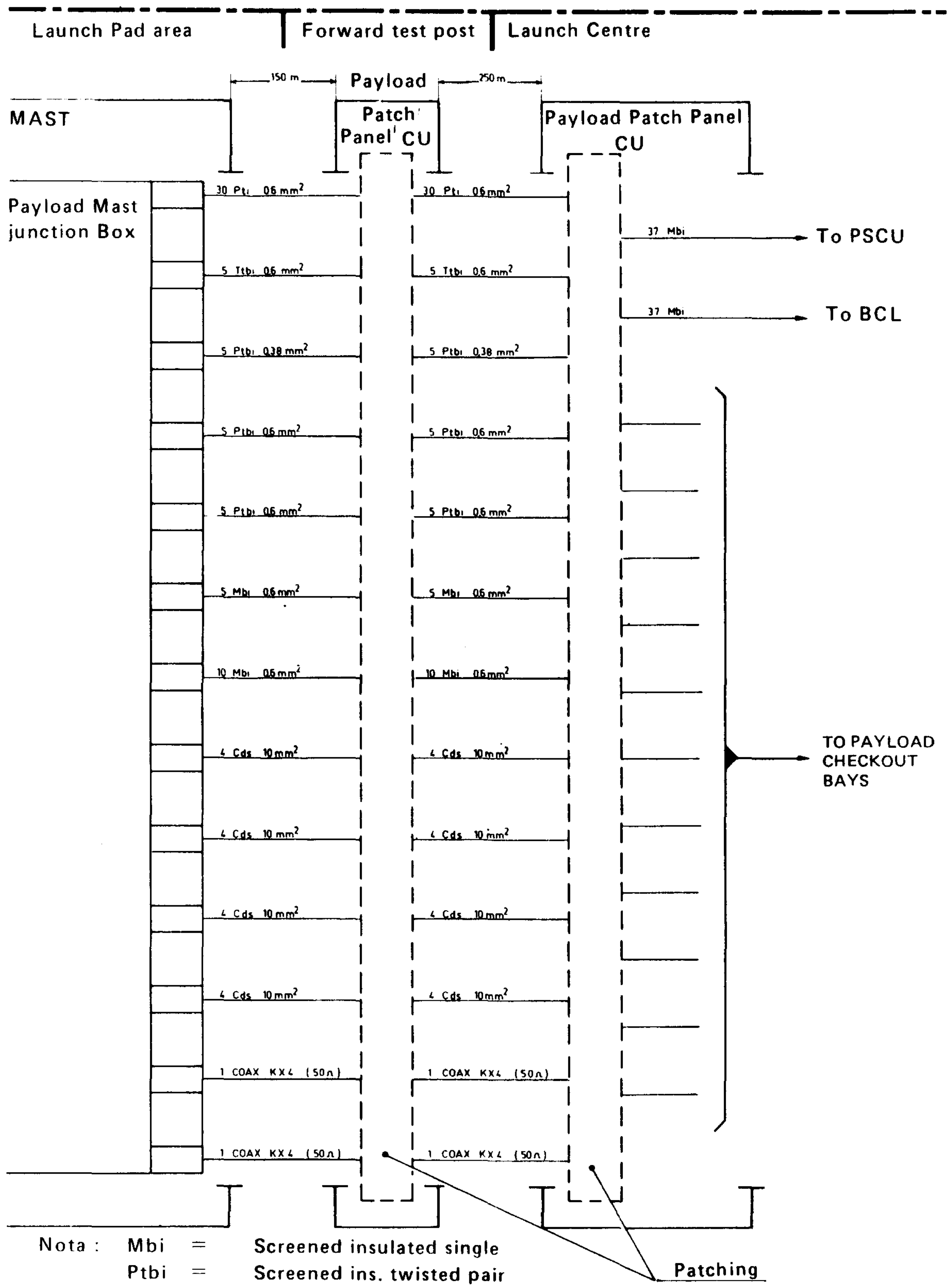
- transmission by the vehicle checkout system of a maximum of 10 commands (for controlling relays or completing circuits) at times to be defined with the user, and using a voltage supplied by the payload checkout system.
- reception by the vehicle checkout system of a maximum of 10 status signals (go-no-go), generated by the payload checkout racks, and scanned by the vehicle checkout system at times to be defined with the user.

The vehicle checkout system distributes universal time and countdown to the payload checkout racks in the Launch Centre.

3.5.8. Radiotransparency of fairing

In the standard version of the fairing, the boat-tail section is made of dielectric material, thus providing radio transparency on the ground and in flight, for payloads having an antenna system radiating through the lower part of the fairing.

The insertion loss of the material used in the boat-tail section is less than 2 dB in the 136-150 MHz and 2100 - 2300 MHz frequency bands.



Nota : Mbi = Screened insulated single
Ptbi = Screened ins. twisted pair
Ttbi = Screened ins. twisted triplet
Pti = Insulated twisted pair
Cds = Insulated Conductor

FIG. 3.41 PAYLOAD / LAUNCH CENTRE CABLES

According to the configuration of the antenna systems used by the payload, additional studies can be undertaken and the following options are possible :

- location of radiotransparent windows in the area authorized for the fitting of access doors in the cylindrical section (see figs. 3.22a and b) of the fairing, or location of transfer antennas on the fairing skin,
- for ground checkout of the payload, location of radiofrequency repeaters near the fairing, to eliminate, if necessary, the masking effect of the servicing tower on transmissions between payload and payload checkout system.

The design of the set-up requested by the user may be verified with the aid of a 1/5-scale satellite and fairing mockup.

3.5.9. Payload installation for launch phase

During the final preparation phase leading up to actual launch, the user must so install his payload that the umbilical cables are only carrying very low currents at the moment of lift-off (for example less than 150 mA at 30 V for a resistive circuit). In particular, battery charging must be stopped several minutes (for example 8 min.) before lift-off. Where ground power supplies are used, these must be cut off at latest 1 minute before launch.

During the powered phase of the launch vehicle and up to separation of the payload(s), no telecommand signal can be sent to the payload(s), or generated by an on-board system (sequencer, computer, etc.). To command operations on the payload immediately after separation from the launch vehicle, microswitches or telecommand systems can be used. Initiation of operations on the payload after separation from the launch vehicle, by an on-board payload system, preprogrammed before lift-off, is not authorized.

3.5.10 Radio compatibility

In order to ensure the radio compatibility of the launch vehicle and payload, a frequency plan will be drawn up for each launch. The user will be required to supply all data needed for preparation of this plan (see para. 4.4.9 of the " Application to use Ariane ", reproduced in para. 6.4. of this Manual).

The launch vehicle is equipped with the following transmission and reception systems :

- A telemetry system with the transmitter in the VEB and an antenna system comprising two antennae located on the cylindrical section of the bay, having an omnidirectional radiation pattern and no special polarization. The transmission frequency is in the 2200 - 2290-MHz band, and the transmitter power is 20 W.
- A telecommand-destruct reception system, comprising two receivers operating in the 420 - 450-MHz band. Each receiver is coupled to a system of two antennae, located on the VEB, having an omnidirectional pattern and no special polarization.

• A radar transponder system, comprising two identical transponders with a reception frequency of 5690 MHz, and transmission frequencies in the 5400 - 5900-MHz band. The minimum transmitting power of each transponder is 400 W peak. Each transponder is coupled to a system of two antennae, located on the VEB, with an omnidirectional pattern and clockwise circular polarization.

Any spurious radiation from the launch vehicle transmission systems is limited so that electrical fields 0.5 m above the payload separation plane (using a type 1194 adaptor) do not exceed the following limits :

band :	discrete frequencies :	in wide band :
148 to 150 MHz 2024 to 2121 MHz	15 dBμV/m 35 dBμV/m	25 dBμV/m/MHz ½ 40 dBμV/m/MHz ½

Spurious radiation from the launch vehicle to the ground is limited to − 55 dBW on discrete frequencies in the frequency bands normally used by payloads for satellite-to-ground links.

Spurious radiation from the payload must not produce electrical fields exceeding the following limits, on the VEB antennae (1 m below the payload separation plane) :

In the telecommand-receiver 420 - 450 MHz band round its working frequency f :

f − 0.5 MHz to f + 0.5 MHz :	20 dBμV/m measured with a bandwidth of 200 kHz.
f − 1 MHz to f − 0.5 MHz f + 0.5 MHz to f + 1 MHz	35 dBμV/m measured with a bandwidth of 100 kHz
f − 2 MHz to f − 1 MHz f + 1 MHz to f + 2 MHz	55 dBμV/m measured with a bandwidth of 100 kHz
f − 3 MHz to f − 2 MHz f + 2 MHz to f + 3 MHz	70 dBμV/m measured with a bandwidth of 100 kHz.

In the radar-transponder band round its reception frequency f (5690 MHz) :

f − 50 MHz to f + 50 MHz :	70 dBμV/m measured with a bandwidth of 20 MHz.

The levels of discrete spurious radiation transmitted by the payload to the ground must not exceed − 55 dBW in the launch-vehicle transmission bands.

According to user requirements, special studies and, where appropriate, radio-compatibility tests can be undertaken, for the purpose of agreeing a waiver to the above specifications.

3.5.11. Electromagnetic compatibility

An electromagnetic-compatibility study can be made for each payload. For this purpose, the user will supply the necessary data, and in particular the standards that he applies (see para. 4.4.10. of the " Application to use Ariane ").

The launch vehicle meets the following electromagnetic-compatibility requirements :

- The launch vehicle radiates no interference having a level exceeding those given in figures 3.42, 3.43 and 3.44.

  These levels are measured at 0.5 m above the payload separation plane (using a type 1194 adaptor), and do not take account of intentional transmission by the launch vehicle.

  - Figure 3.42 : Spurious radiation by launcher : narrow-band electrical field
  - Figure 3.43 : Spurious radiation by launcher : wide-band electrical field (coherent noise)
  - Figure 3.44 : Spurious radiation by launcher : narrow-band magnetic field

- The payload does not radiate a narrow-band electrical field in the VEB (0.5 m below the payload separation plane) exceeding the limit set in figure 3.45 (including intentional transmission).

The electromagnetic environment of the CSG is defined in the CSG Manual.

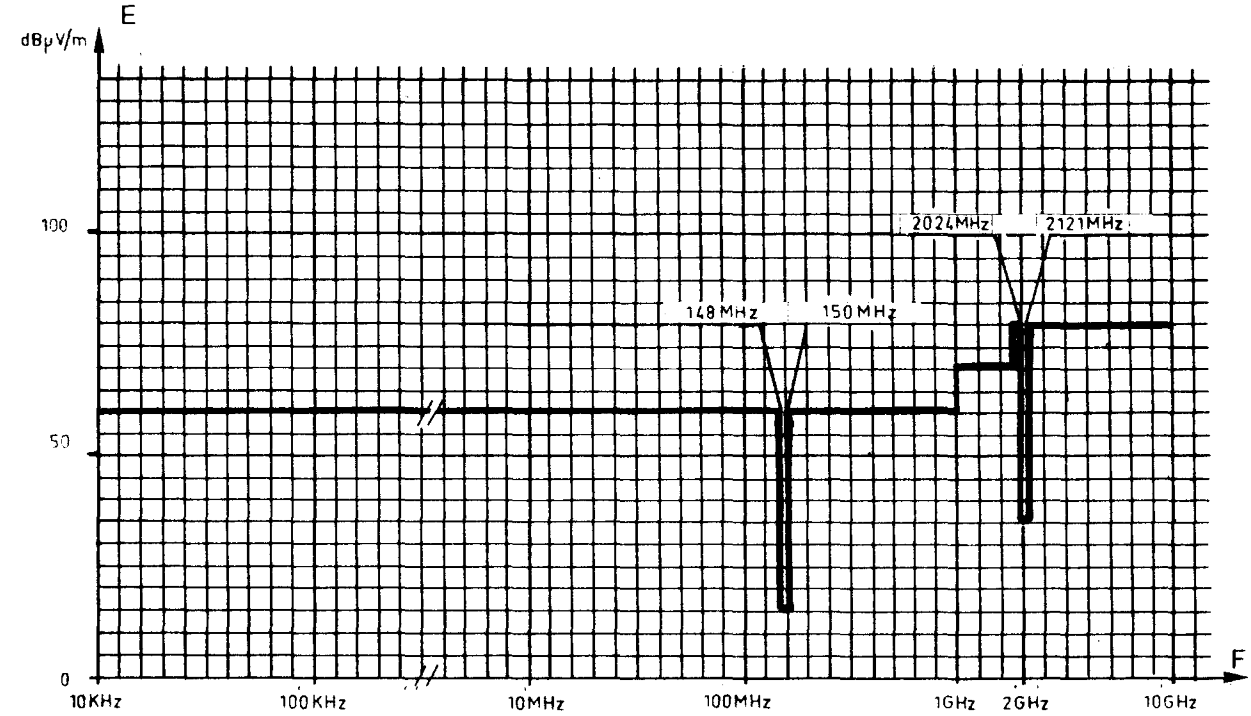


FIG 3.42 SPURIOUS RADIATION BY LAUNCHER
NARROW–BAND ELECTRICAL FIELD

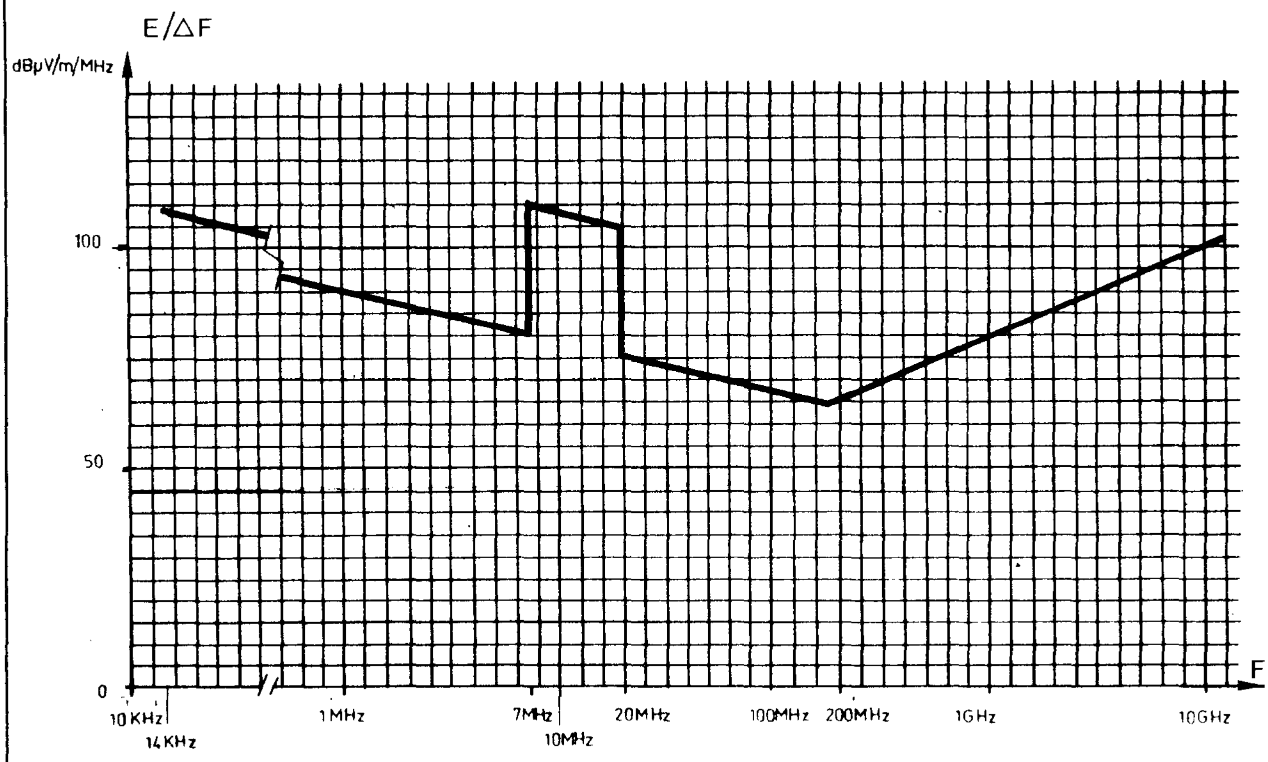


FIG. 3.43 SPURIOUS RADIATION BY LAUNCHER
WIDE–BAND ELECTRICAL FIELD (COHERENT NOISE)

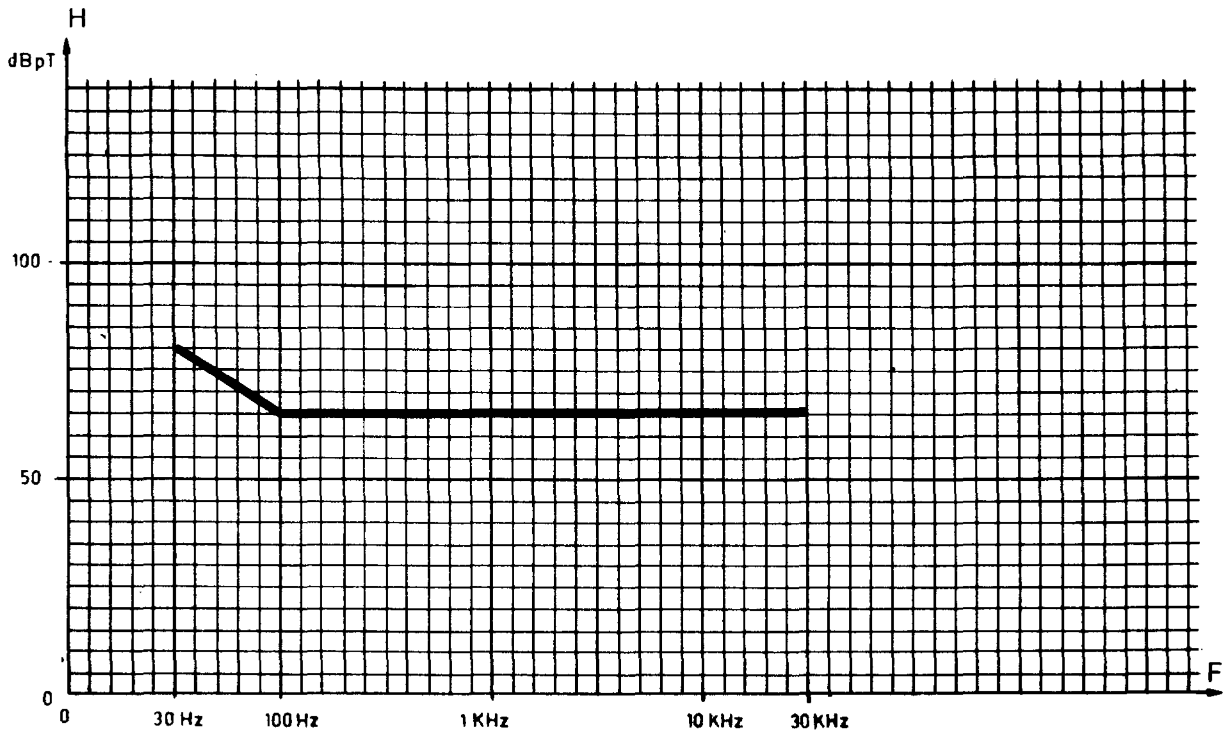


FIG. 3.44 SPURIOUS RADIATION BY LAUNCHER
NARROW–BAND MAGNETIC FIELD

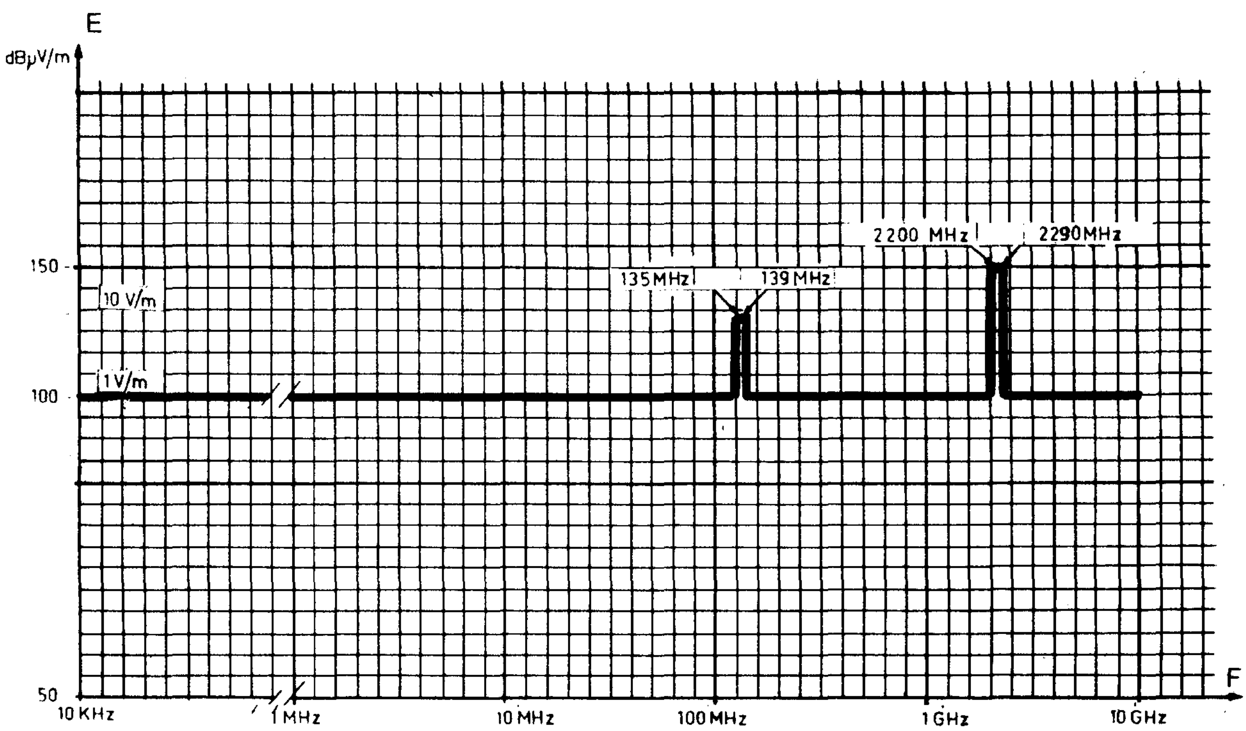


FIG. 3.45 SPURIOUS RADIATION ACCEPTABLE TO LAUNCHER
NARROW–BAND ELECTRICAL FIELD

Launch operation

Chapter 4

4.1. Introduction

4.1.1. General

Ariane launches will be made from the Guiana Space Centre (CSG) at Kourou in French Guiana.

This chapter describes payload operations to be executed at the CSG (para. 4.2., 4.3. and 4.4.), and the facilities available to users (para. 4.6., 4.7. and 4.8.).

Maps of the CSG and the sites concerned with operations are shown in figures 4.1, 4.2, 4.3 and 4.4.

General information concerning French Guiana is given in Annex 2.

A detailed description of the CSG and associated equipment will be found in the " CSG Manual ".

The CSG management is responsible to the Ariane authority for services relating to launch operations, as defined in the launch order (see chapter 6).

4.1.2. Definition of operations

Operations at the CSG are executed in three phases :

- Phase 1 : Preparation and checkout of satellite
- Phase 2 : Hazardous operations
- Phase 3 : Operations in the launch zone

Phases 1 and 2 cover operations carried out in the Payload Preparation Complex (EPCU), and are dissociated from those specific to the launch-vehicle.

Phase 3 concerns combined payload/launch-vehicle operations in the Ariane launch zone.

Payload operations are covered by a satellite operations plan, prepared by the user. Procedures classified as " hazardous " are subject to approval by the CSG Safety Department (see chapter 5).

The operational organization of a launch compaign is detailed in para. 4.5.

(1)– ARIANE LAUNCH SITE (ELA)*

(2)– STATION KRU 92 : TM – TC (IRIS)
INTERFEROMETRY (DIANE)

(3)– STORAGE (MISCELLANEOUS EQUIPMENT)

(4)– STORAGE (MOTORS AND
PYROTECHNICS : PR 3 and PY3)

(5)– CSG TECHNICAL CENTRE
(S1, control centre)

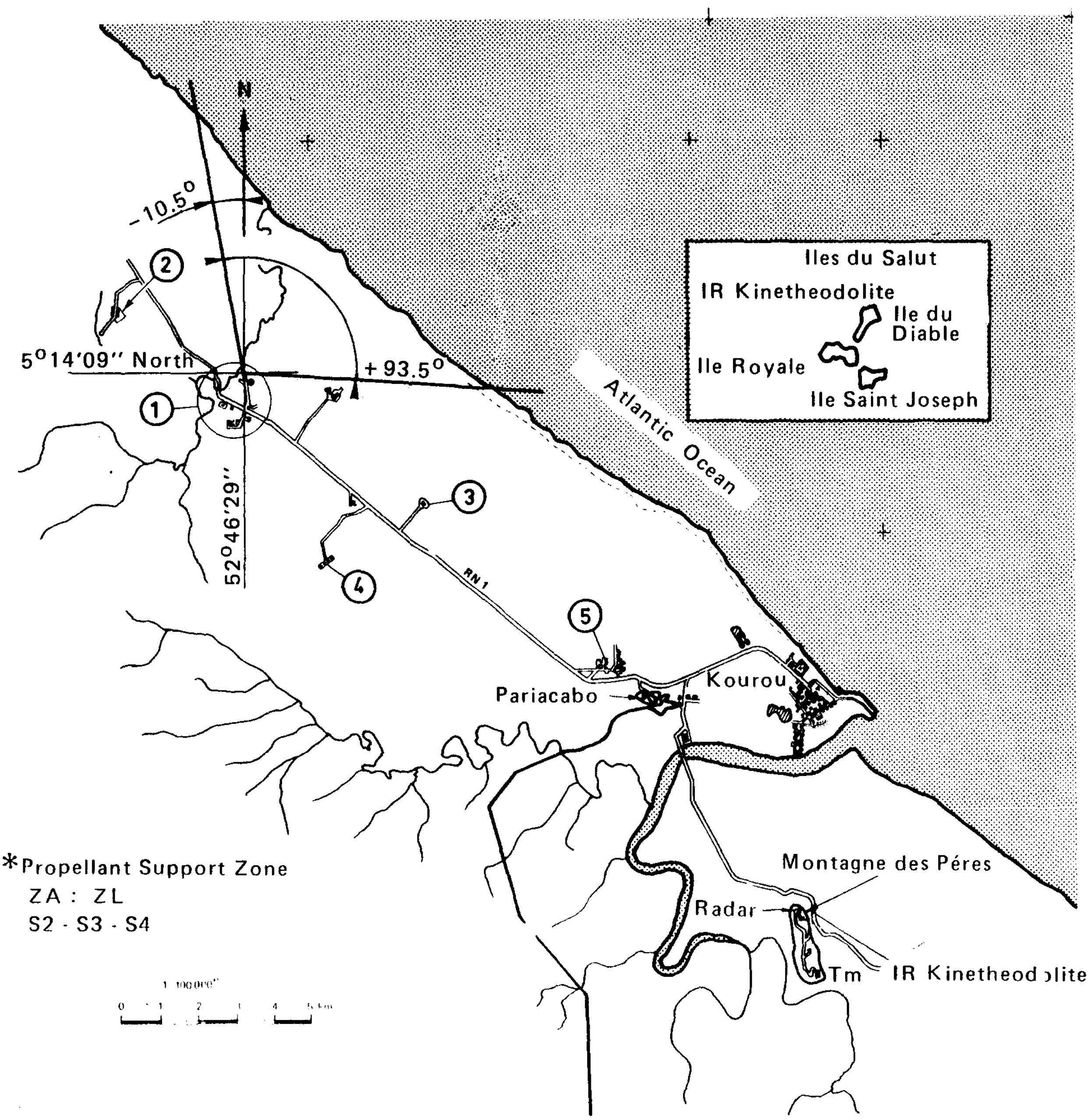


FIG. 4.1 GUIANA SPACE CENTRE (CSG)

(1) CHECK POINT
(2) OFFICES (URANUS, NEPTUNE)
(3) OPTICS SECTION VIEWING ROOM (SATURNE)
(4) CONTROL CENTRE (JUPITER)
(5) CONTROL CENTRE ANNEX (MARS)
(6) TELECOM' CENTRE (MERCURE)
(7) SPACECRAFT PREPARATION S1
(8) FIRE BRIGADE
(9) GENERAL STORE
(10) OPEN STORAGE
(11) ELECTROMECHANICAL WORKSHOP
(12) GARAGE – FILLING STATION
(13) HELICOPTER PLATFORM

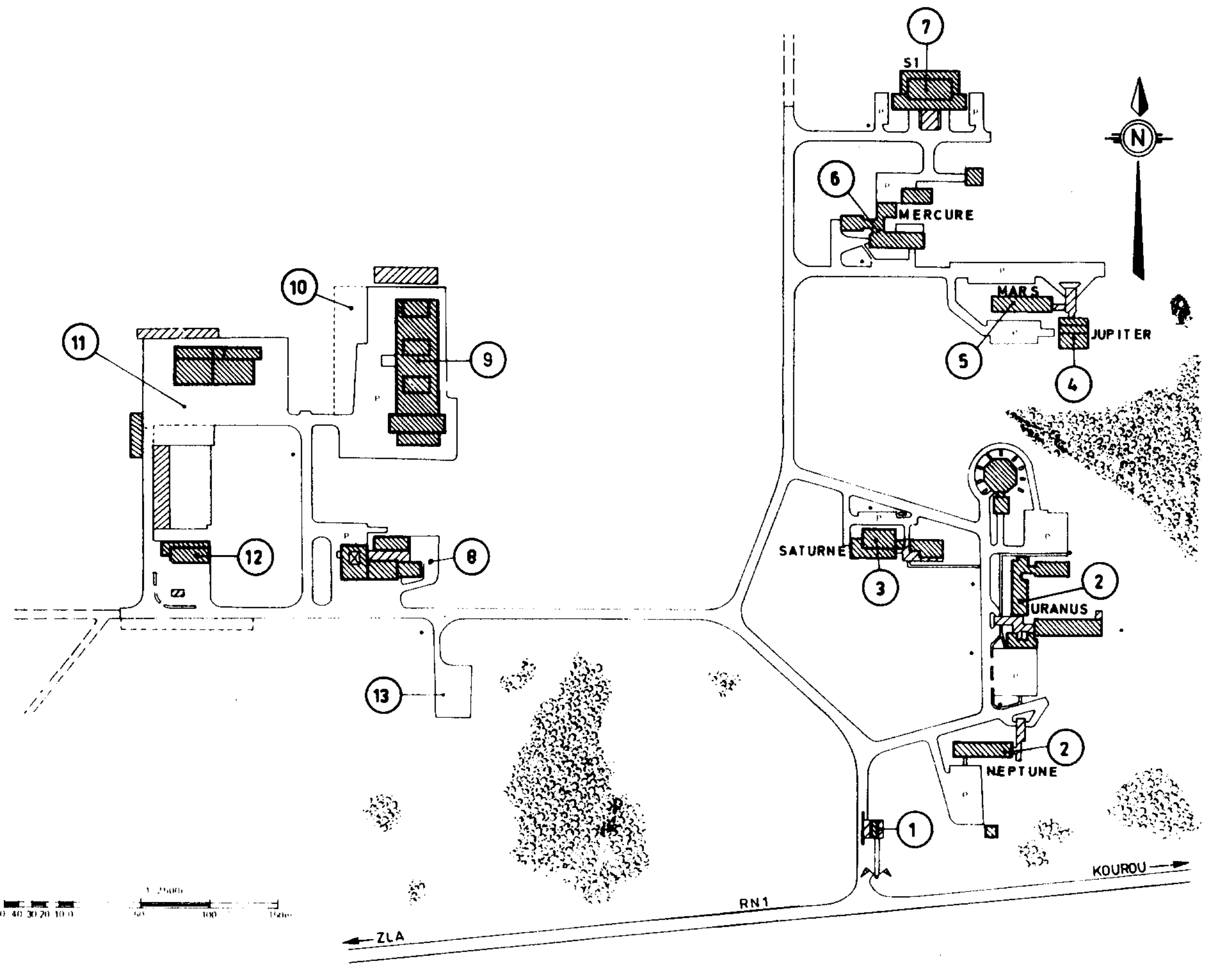


FIG. 4.2 CSG TECHNICAL CENTRE

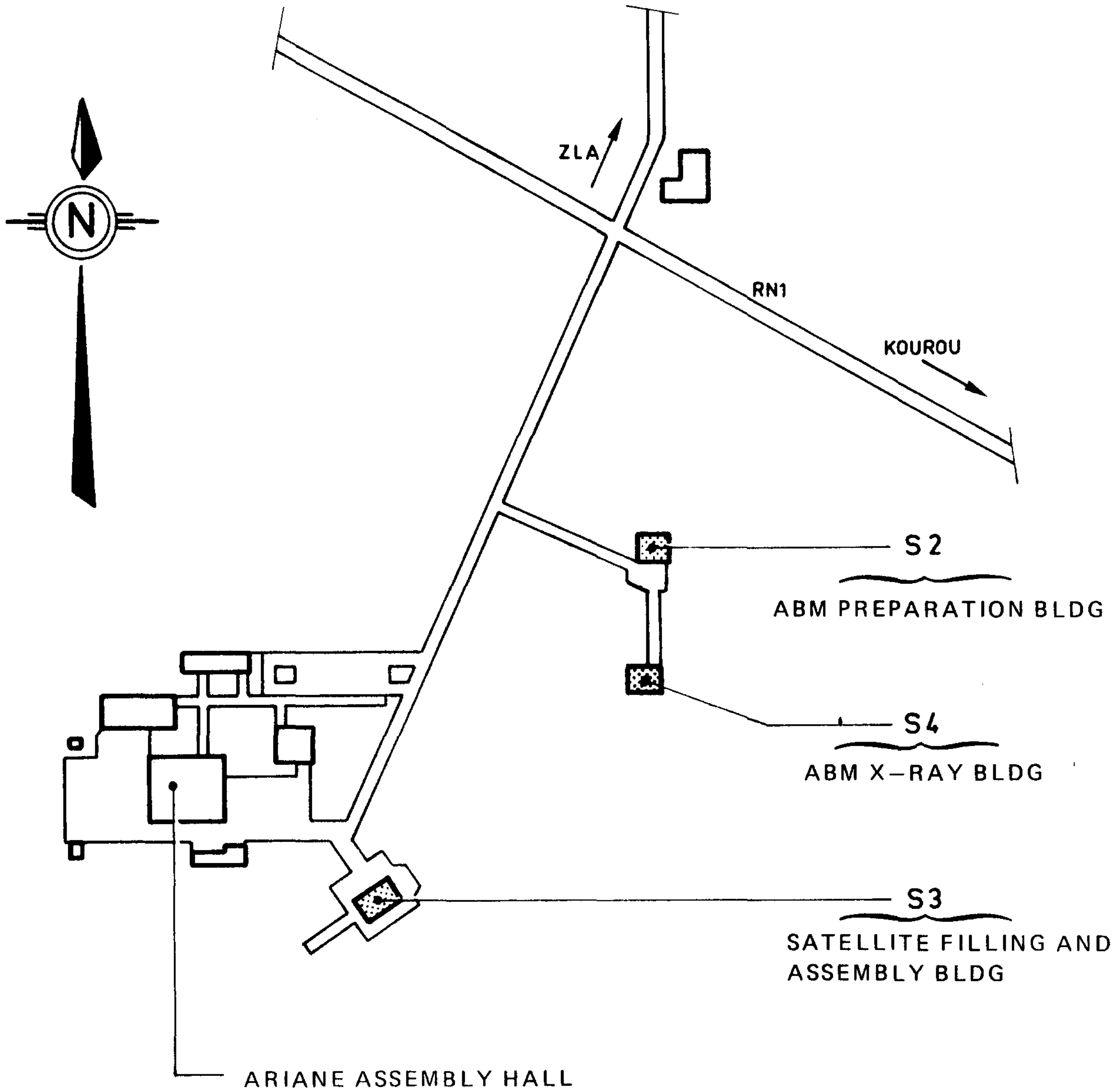
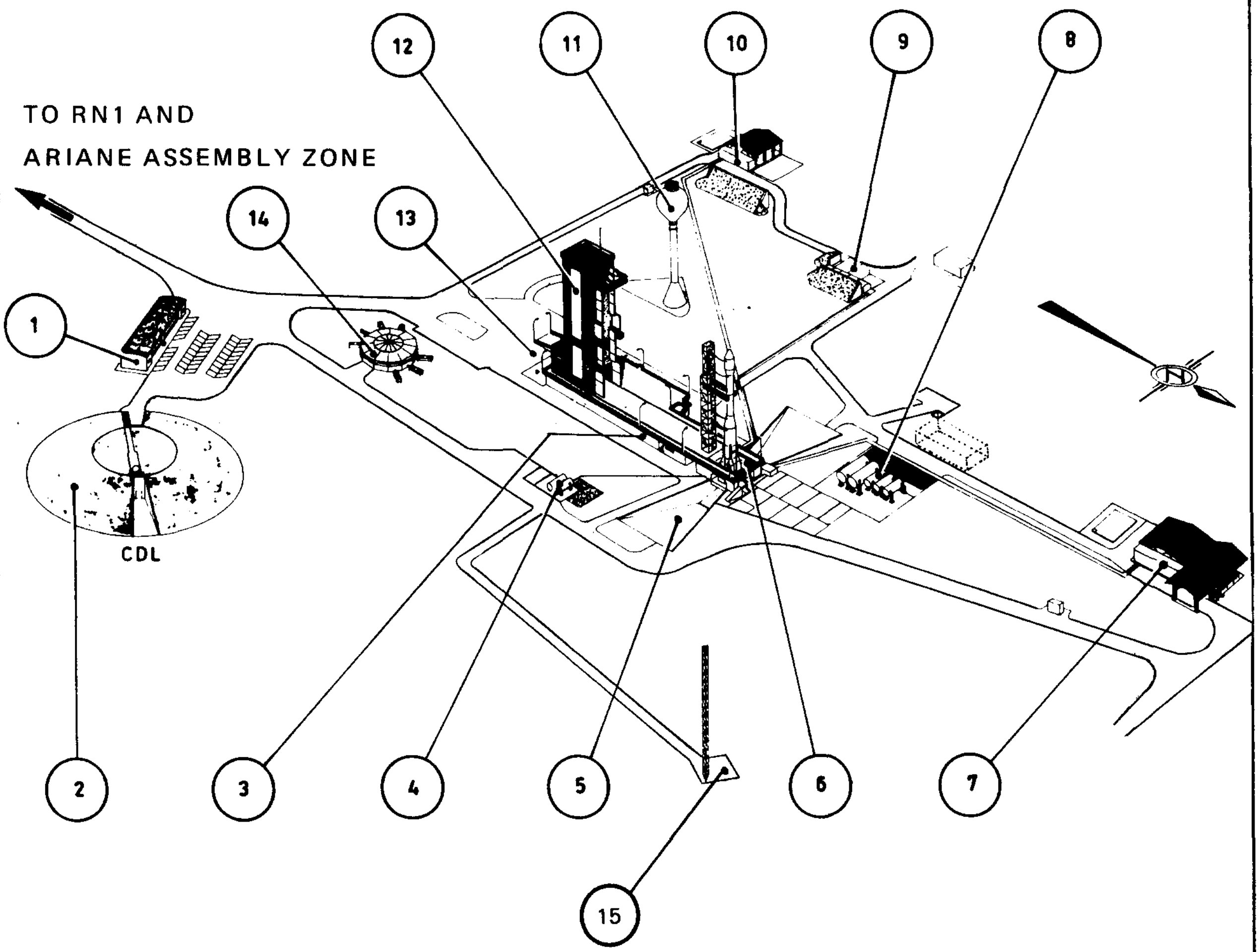


FIG. 4.3 ARIANE ASSEMBLY ZONE – ZA



1 SUPPORT SERVICES BUILDING	8 NITROGEN STORAGE
2 LAUNCH CENTRE	9 LIQUID HYDROGEN STORAGE
3 FORWARD TEST POST	10 UDMH STORAGE
4 LIQUID OXYGEN STORAGE	11 WATER TOWER
5 FLUES	12 SERVICING TOWER (WITHDRAWN)
6 LAUNCH TABLE	13 HELIUM STORAGE
7 N_2O_4 STORAGE	14 ICED WATER PLANT
	15 METEO MAST

FIG. 4.4 ARIANE LAUNCH ZONE – ZL

4.1.3. Block diagram of operations

A typical block diagram is given in figure 4.5., for guidance purposes.

4.1.4. Typical time-schedule of operations

A typical time-schedule is given in figure 4.6., for guidance purposes.

4.2. Phase 1 - Preparation and checkout of satellite

4.2.1. Unloading in Guiana

Details of port and airport facilities are given in para. 2.5 of Annex 2, and are also described in the CSG Manual. Unloading is carried out by the port or airport authorities as appropriate.

4.2.2. Transport to the CSG

The port and airport of Cayenne are linked to the CSG by highways RN1 and RN2. The CSG is responsible for transport within Guiana.

4.2.3. Off-loading at S1

On arrival at the CSG, the satellite in its container, together with associated ground equipment, are taken to the technical centre and off-loaded in the stores areas of building S1.

In the satellite operations plan, the user proposes how his facilities should be arranged and laid out in this building.

4.2.4. Storage

The ABM in its container is stored in building PR 3. The pyrotechnic systems, radioactive sources and any other hazardous system of the same class are stored in building PY 3. Hazardous fluids are stored in the propellant-support zone of the Ariane launch site.

The user states what equipment must be stored in an air-conditioned environment. Other equipment will be stored under cover or in the open air.

In the satellite operations plan, the user proposes how his facilities should be arranged and laid out in the buildings.

4.2.5. Installation

Ground check-out equipment and the satellite are installed in building S1 as follows :

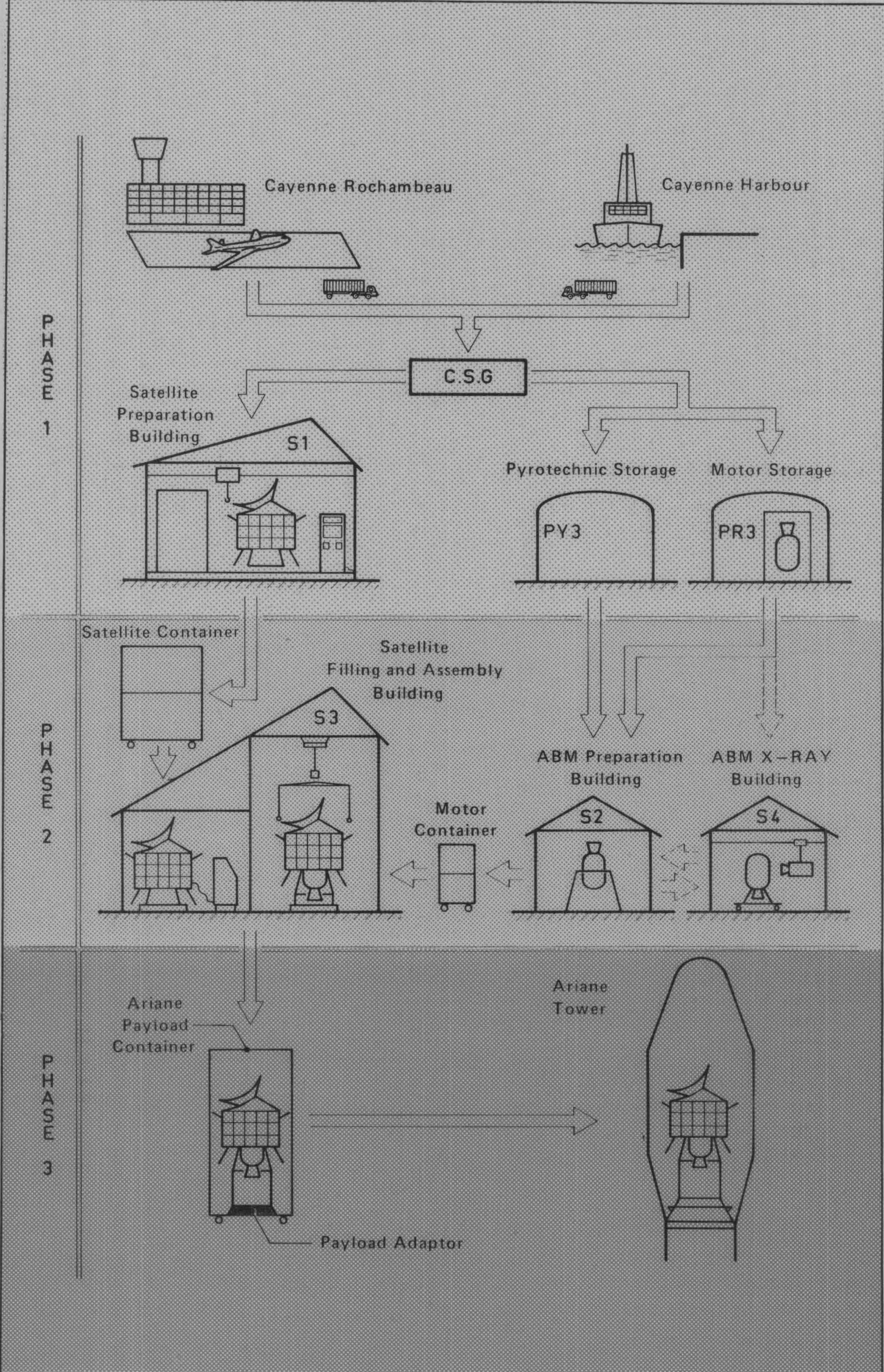


FIG. 4.5 OPERATIONS : BLOCK DIAGRAM

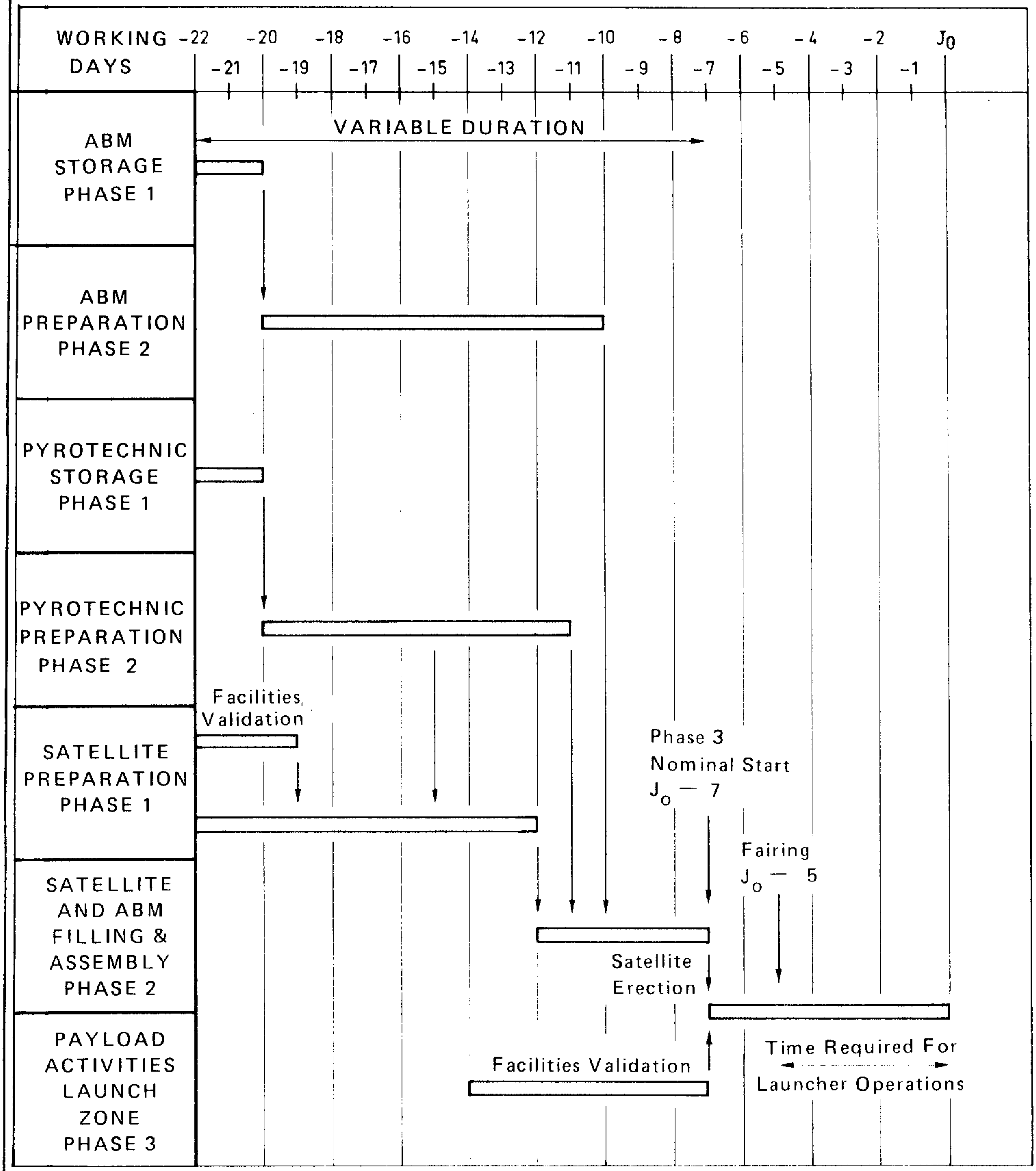


FIG. 4.6 TYPICAL OPERATIONS TIME–SCHEDULE

- ground equipment : satellite checkout equipment is installed , connected to CSG facilities, and validated.
- satellite : the satellite is extracted from its container, and installed in the clean zone. This can also apply to replacement flight equipment.

4.2.6. Preparation and checkout

The satellite is assembled and undergoes a functional check (non-hazardous mechanical and electrical tests). The satellite is complete, but without ABM and empty of propellants.

4.2.7. Packing

When all checks have been completed on the satellite, the latter is placed in its own container or the payload container (CCU), to await transport to the Ariane Assembly Zone.

4.2.8. Block-diagram of Phase 1 operations

See figure 4.7.

4.3. Phase 2 - Hazardous operations

4.3.1. General

Hazardous operations are carried out in buildings S2, S3 and S4 for the solid-propellant ABM, and building S3 for the satellite. The liquid-propellant engines are prepared in the filling hall in building S3. Validation of certain satellite ground equipment and the filling and pressurization system is carried out in building S3 before arrival of the satellite. The satellite and the ABM are assembled in the assembly hall in building S3.

Certain Phase 2 operations can be undertaken in parallel with Phase 1 operations. The order and number of operations to be executed can differ, according to the specific characteristics of each satellite.

4.3.2. Pyrotechnic preparation

Checks of the pyrotechnic systems required before assembly on the satellite are carried out in building S3.

4.3.3. Preparation of radioactive sources

Radioactive sources that must be checked before assembly on the satellite are taken from building PY3 to building S3.

4.3.4. Preparation of ABMs

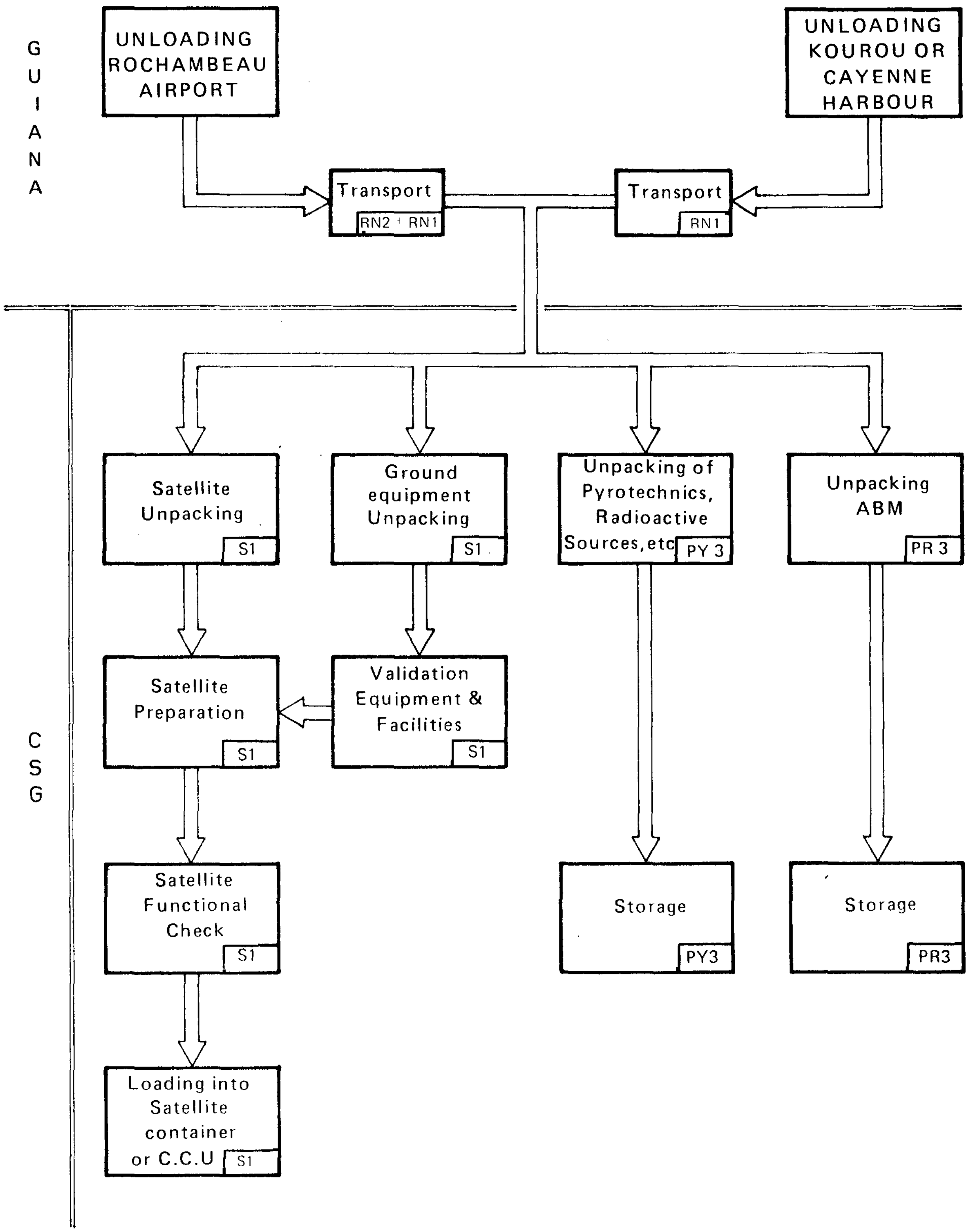


FIG. 4.7 OPERATIONS – PHASE 1 : BLOCK DIAGRAM

4.3.4.1. Removal from store and transport to S2

ABMs stored in building PR3 can undergo X-ray examination in building S4, before or after transport in their containers to building S2.

4.3.4.2. Preparation and checks

The ABM is assembled and checked. In particular, the pyrotechnic motor-ignition system is transported from building PY3 to building S2, for checking prior to assembly with the motor.

4.3.4.3. Packing and transport to S3

The ABM, assembled with its igniter, is placed in its container and transported to the access airlock of building S3.

4.3.4.4. Preparation for assembly

The ABM and igniter are extracted from the container, and placed in the assembly hall in building S3.

4.3.5. Operations on the satellite

4.3.5.1. Transport and installation at S3

The satellite in its container is transported from building S1 to the access air lock of building S3. It is removed from its container, and mounted on its handling trolley in the filling hall.

4.3.5.2. Fluid filling and pressurization

The satellite is prepared for fluid filling and partial or complete pressurization. Leak tests are carried out a regular intervals, and the satellite is then transported to the assembly hall.

The satellite batteries can be charged in S3, except during hazardous operations.

In case of need, depressurization, purging and flushing operations can be carried out in the filling hall.

4.3.5.3. Assembly of pyrotechnics, radioactive sources and miscellaneous items.

Assembly of various sources of danger (pyrotechnic devices, radioactive sources, etc.) on the satellite is carried out in the assembly hall.

4.3.6. Assembly of satellite with ABM

4.3.6.1. Assembly

The ABM is integrated with the satellite, the resultant assembly forming the payload (" Charge Utile " - CU).

4.3.6.2. Balancing and weighing

The payload is balanced if necessary. The balancing operation is carried out in the filling hall, on the satellite with or without its ABM and fluids.

The payload is then weighed in the assembly hall.

4.3.6.3. Checks and inspection

Electrical, mechanical and arming checks can be carried out in the assembly hall. A final inspection is made before placing the payload in its container.

4.3.6.4. Placing of payload in its container

This operation is carried out by the Ariane authority, under the control of the payload authority.

4.3.7. Block diagram of Phase 2 operations

See figure 4.8.

4.4. Phase 3. Operations in the launch zone

4.4.1. General

The launch zone comprises the launch-pad area, on which the launch vehicle and payload are located, and the Launch Centre.

In the launch zone, operations on the payload are linked with those on the launch vehicle. A detailed time-schedule for combined launch-vehicle/payload operations is given in the Launch Order. This schedule may be revised on the arrival of the satellite at the CSG, and on completion of Phase 2 operations.

Validation operations on the payload ground equipment are carried out — on the servicing tower and in the Launch Centre — before the payload is mounted on the launch vehicle.

The clean area at the top of the tower (PF8) is reserved exclusively for payload operations, up to the time of fitting the fairing.

During the launch rehearsal and during countdown, the payload preparation team must be available for activating and checking the payload, which form part of the combined launch-vehicle/payload operations.

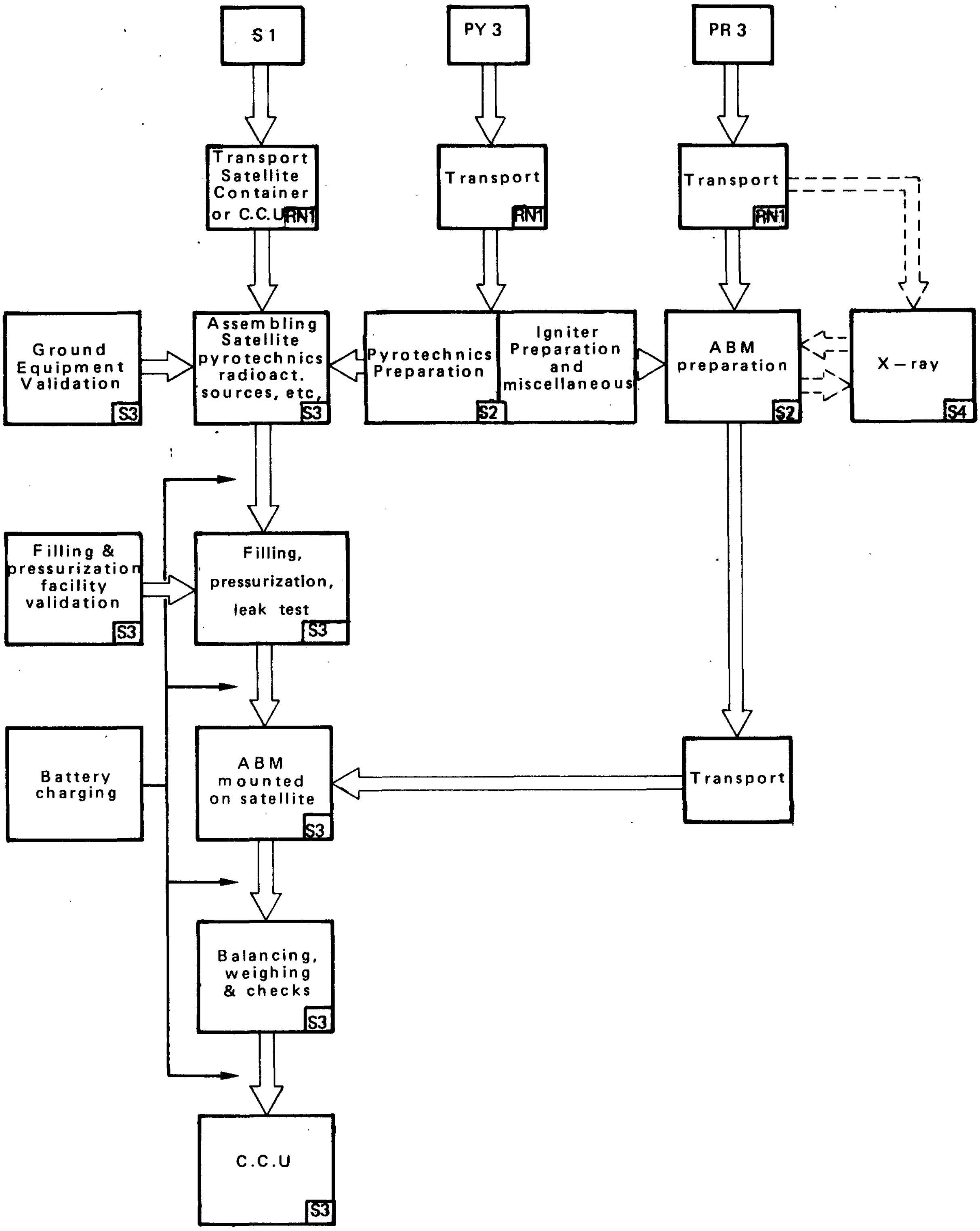


FIG. 4.8 OPERATIONS – PHASE 2 : BLOCK DIAGRAM

4.4.2. **Transport of payload**

The Ariane authority is responsible for transporting the payload from building S3 to the servicing tower, using the CCU and its trolley. After uncoupling from the platform at the foot of the tower, the CCU is raised to level PF8, transferred to the clean-area airlock, and lowered on to this platform as shown in figure 4.9. When nominal conditions have been re-established in the airlock (cleanliness, temperature and relative humidity), the CCU is decontaminated and opened.

4.4.3. **Assembly on the launch vehicle**

When the CCU has been opened, the payload preparation team fits the handling device on the payload. When the mechanical link between the CCU and the base of the payload has been uncoupled, the payload is raised, moved towards the launch vehicle using the PF8 clean-area hoist, and lowered on to the payload flight adaptor (see fig. 4.10). The completion of the mechanical link between payload and adaptor, by the Ariane team, authorizes removal of the payload-handling device by the payload preparation team. The CCU is then removed.

The payload is then connected electrically to the umbilical mast :

- automatically via the launch-vehicle equipment bay ;
- via an extension cable to the payload mast junction box, in cases where an umbilical link via the fairing is specified (see para. 3.5.5.).

4.4.4. **Preparation and checkout of payload.**

Preparation and checkout of the payload must allow for accessibility and radio-silence constraints.

An " inert " payload has no active sub-system. A payload " on standby " is defined as presenting a steady state (no mechanical or electrical change of state), with no radio transmission. Trickle charging of batteries is admissible, and radio receivers may be permanently switched on. These conditions are discussed at the time of the safety submissions, as described in chapter 5, and adjusted when finalizing the Combined Operations Plan (POC).

Electrical cabling (continuity, insulation, etc.) and radio links are checked.

A functional check of the payload is carried out in accordance with the combined activities time-schedule, together with certain routine operations (leak testing, state of charge of batteries, visual inspection).

Arming and disarming checks of hazardous circuits are carried out with the approval of the Launch Centre safety officer.

4.4.5. **Radio compatibility and pressurization**

A launch-vehicle/payload radio compatibility check is made, if the payload is to be launched with its transmission/reception systems operating.

LEGEND 1 - HOISTING CCU
2 - ENTERING PF 8 AIRLOCK
3 - SETTING DOWN CCU ON PF8

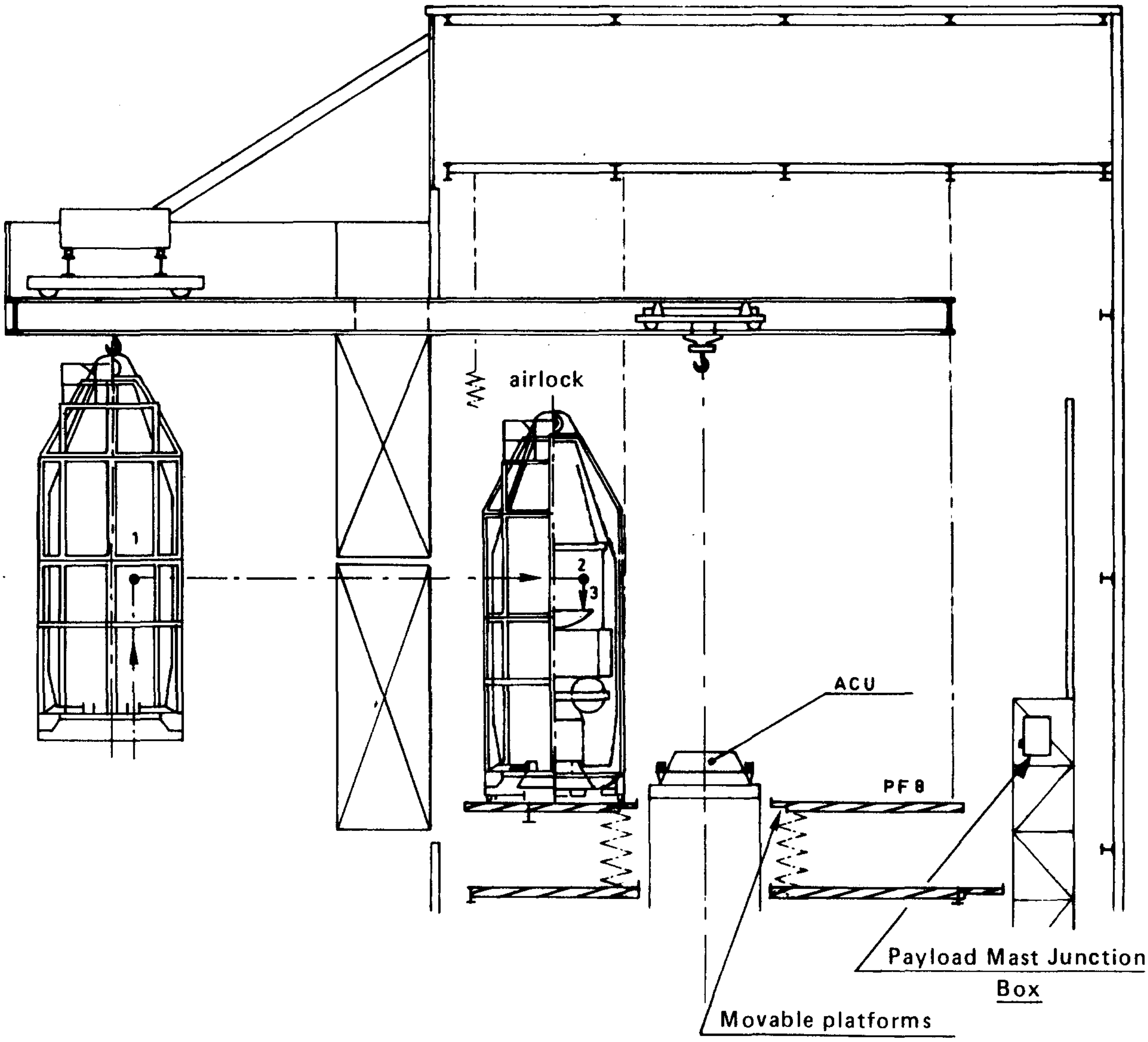


FIG. 4.9 SERVICING TOWER
INTRODUCTION OF CCU INTO PF 8

LEGEND: 1 - HOISTING PAYLOAD
2 - PAYLOAD TRANSFER
3 - END OF PAYLOAD TRANSFER
4 - SETTING DOWN PAYLOAD ON LAUNCHER

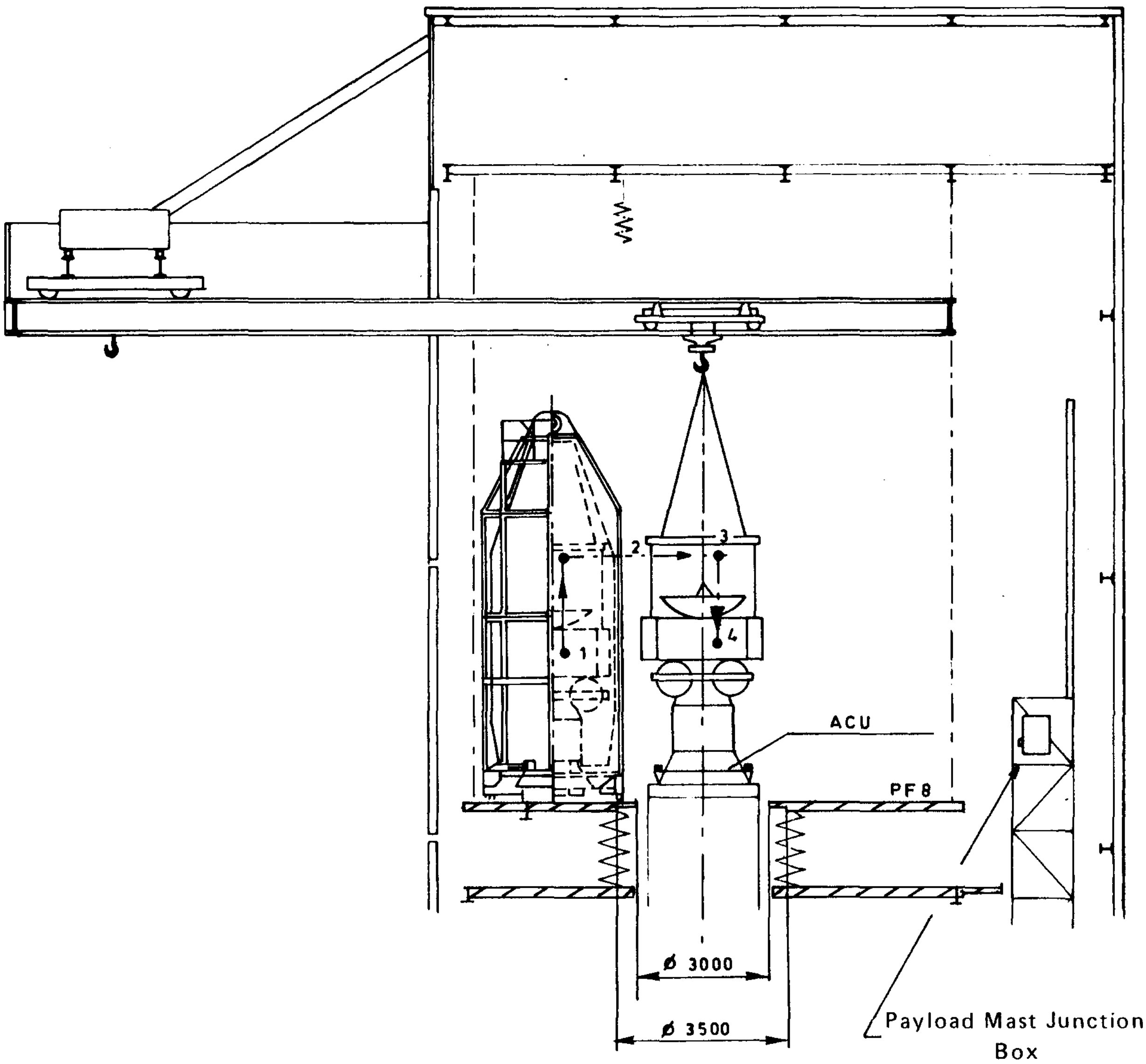


FIG. 4.10 LAUNCH TOWER
ERECTION OF PAYLOAD ON LAUNCHER


ariane
MR
esa
cnes

ARIANE launch complex
(Kourou, French Guiana)

Final pressurization of payload fluids to flight pressure is carried out if this operation was not completed in Phase 2.

4.4.6. Fairing

Preparation of the payload in its flight configuration must be completed before the nose fairing is fitted by the Ariane team. Only simple operations or checks on the payload (connection checks, pyrotechnic-system checks, removal of protective devices, etc.) are possible after the fairing has been fitted, using the access doors provided for this purpose.

4.4.7. Launch rehearsal

A launch rehearsal is held. This includes all electrical and mechanical operations from H0 — 8 hours of the launch countdown with simulated vehicle flight. This implies the operation of all launch facilities, together with the ground tracking network and satellite checkout equipment.

4.4.8. Checkout and preparation before launch countdown

(See time schedule from J0 −7 to J0 −1, para. 4.4.11.1).

The payload can be checked out via cable and/or radio links, at the authorized times. Access to the payload is prohibited, and radio silence is enforced :

- during fitting and connection of the launch-vehicle pyrotechnic devices on J0 −2 (1st-phase arming) ;
- during calibration of the Inertial Guidance Platform on J0 −3.

4.4.9. Launch countdown

Launch countdown starts 20 hours before launch (H0 −20) with the preparation of the launch vehicle for filling. The main constraints applying to payloads are as follows :

- Access to the payload is strictly limited to specially equipped operators. This means that operations to be executed must be simple, using approved procedures, and relate to equipment that is easily accessible via the doors in the fairing.
- Access to the payload is prohibited during filling operations, the payload being on standby (or inert).
- Operation of the payload electrical and radio systems during " 2nd-phase arming A ". This phase consists in checking that no voltage appears on the pyrotechnic interception plugs (PIP), when all payload and launch-vehicle electrical and radio systems are active.
- Payload on standby (or inert) during fitting of the flight PIP ("2nd-phase arming B ").
- Cut-off of battery charging and power supply from the ground before launch : see chapter 3, para. 3.5.9.

The periods to which these constraints apply are shown in the time-schedules for H0 −20 to H0 (see para. 4.4.11.2. and figs. 4.12 and 4.13.).

4.4.10. Block diagram of Phase 3 operations

See figure 4.11.

4.4.11. Time-schedules

4.4.11.1. Time-schedule J0 − 7 to J0 − 1 :

This nominal time-schedule, expressed in working days, covers Phase 3 operations from J0 − 7 to J0 − 1, J0 being the scheduled launch day.

Ariane	Payload
J0 − 7 : • Preparation of launch vehicle and launch site	• Transport of CCU from S3 to Ariane tower authorized • CCU raised to PF8 and opened • Payload mounted on Ariane • Electrical checks (cable and radio) • Hazardous-system arming and dis-arming checkout
J0 − 6 : • Preparation of launch vehicle and launch site • Payload/Ariane radio compatibility	• Functional check • Payload/Ariane radio compatibility • Pressurization and mechanical preparation
J0 − 5 : • Assembly of fairing • Umbilical connections • Pyrotechnic, circuits and sequence checks • Launch Readiness Review	• Connection of umbilical cable • Checkout of umbilical links • Launch Readiness Review
J0 − 4 : • Rehearsal (8 hours)	• Rehearsal in launch configuration
J0 − 3: • Calibration of inertial platform (12 hours)	• Payload not accessible. Payload on standby, battery charging authorized
J0 − 2 : • Final preparation • Installation of flight batteries • Installation of separation rockets • 1st-phase arming	• Checkout and preparation for countdown
J0 − 1 : • Start of countdown at H0 − 20 h	• Start of countdown at H0 − 15 h at earliest

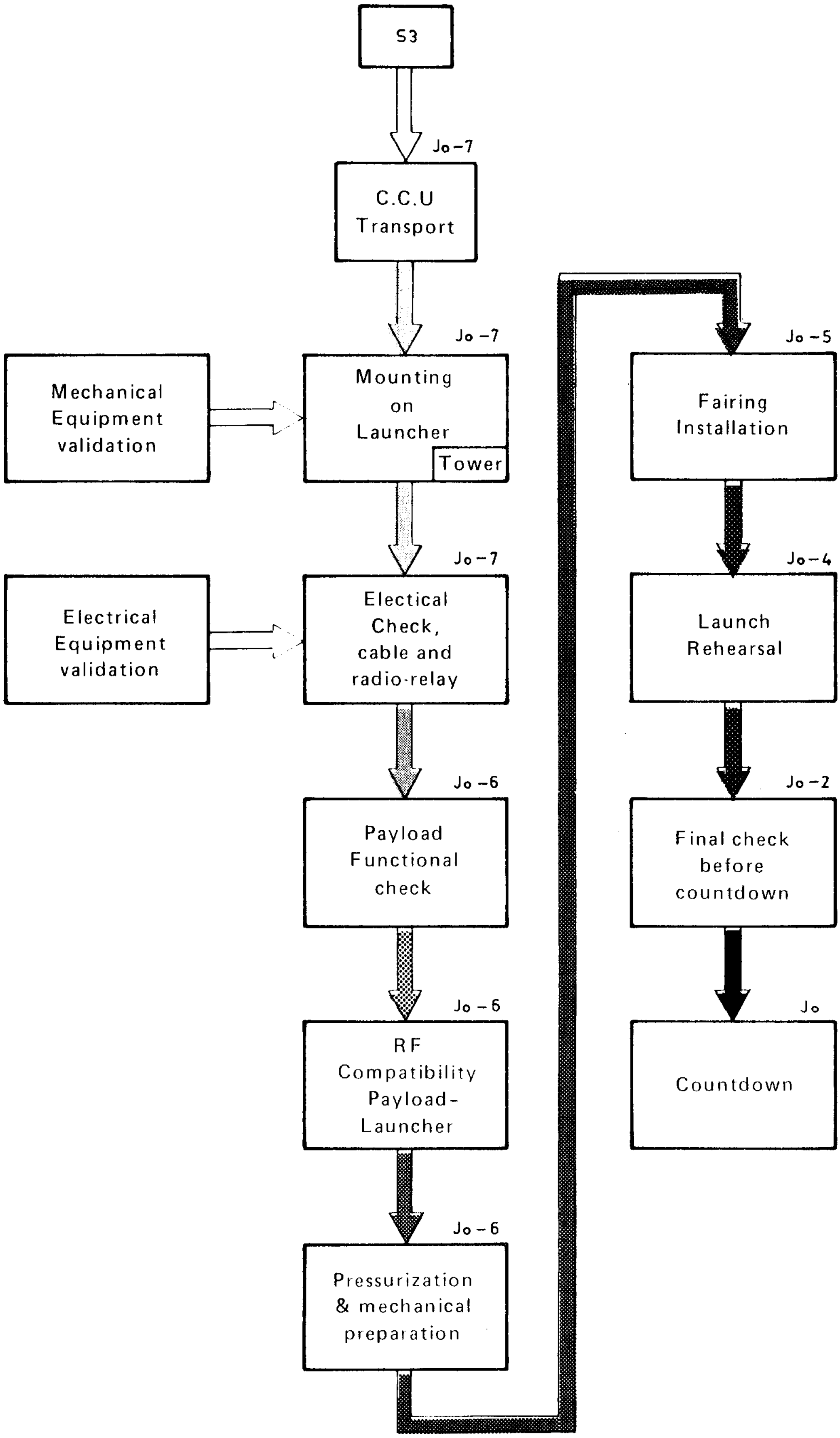


FIG. 4.11 OPERATIONS – PHASE 3 : BLOCK DIAGRAM

4.4.11.2. **Time schedules for H0 – 20 to H0**

See figures 4.12 and 4.13

These nominal time-schedules are expressed in hours, and concern operations from H0 – 20 to H0, H0 being the scheduled launch time.

4.5. Operational organization

The attribution of responsibilities is defined below :

DV : Flight Director (" Directeur de Vol ").
Responsible for checking the compatibility of payload mission objectives with the capability of the launch system. He delegates the preparation and execution of the launch campaign to the Mission Director.

CPCU : Satellite Project Manager (" Chef de Projet Charge Utile ").
Delegates :
1) preparation, activation and checkout of the payload to the Payload Preparation Manager.
2) preparation and execution of the orbit flight plan to the Orbital Operations Manager.

CPA : Ariane Project Manager (" Chef du Projet Ariane ").
Delegates preparation, activation and checkout of the launch vehicle and launch site facilities to the Head of Launch Site Operations.

CSG : Range Director.
Delegates :
1) preparation, activation and operational coordination of the CSG facilities and Ariane down-range stations to the Director of Operations.
2) safety of persons and property to the Safety Officer.

CM : Mission Director (" Chef de Mission ").
Responsible for preparation and execution of the launch campaign.

ACM : Assistant Mission Director (" Adjoint au Chef de Mission ").
Responsible to the Mission Head for operational coordination of payload preparation. He liaises with the CSG Payload Authority, and is responsible for synthesis of reports (payload and payload orbital operations ground station network).

DDO : Launch Operations Director (" Directeur des Opérations ").
Responsible for the preparation, activation and operational coordination of the CSG facilities and Ariane down-range stations.

RS : Safety Officer (" Responsable Sauvegarde ").
Responsible for the safety of persons and property

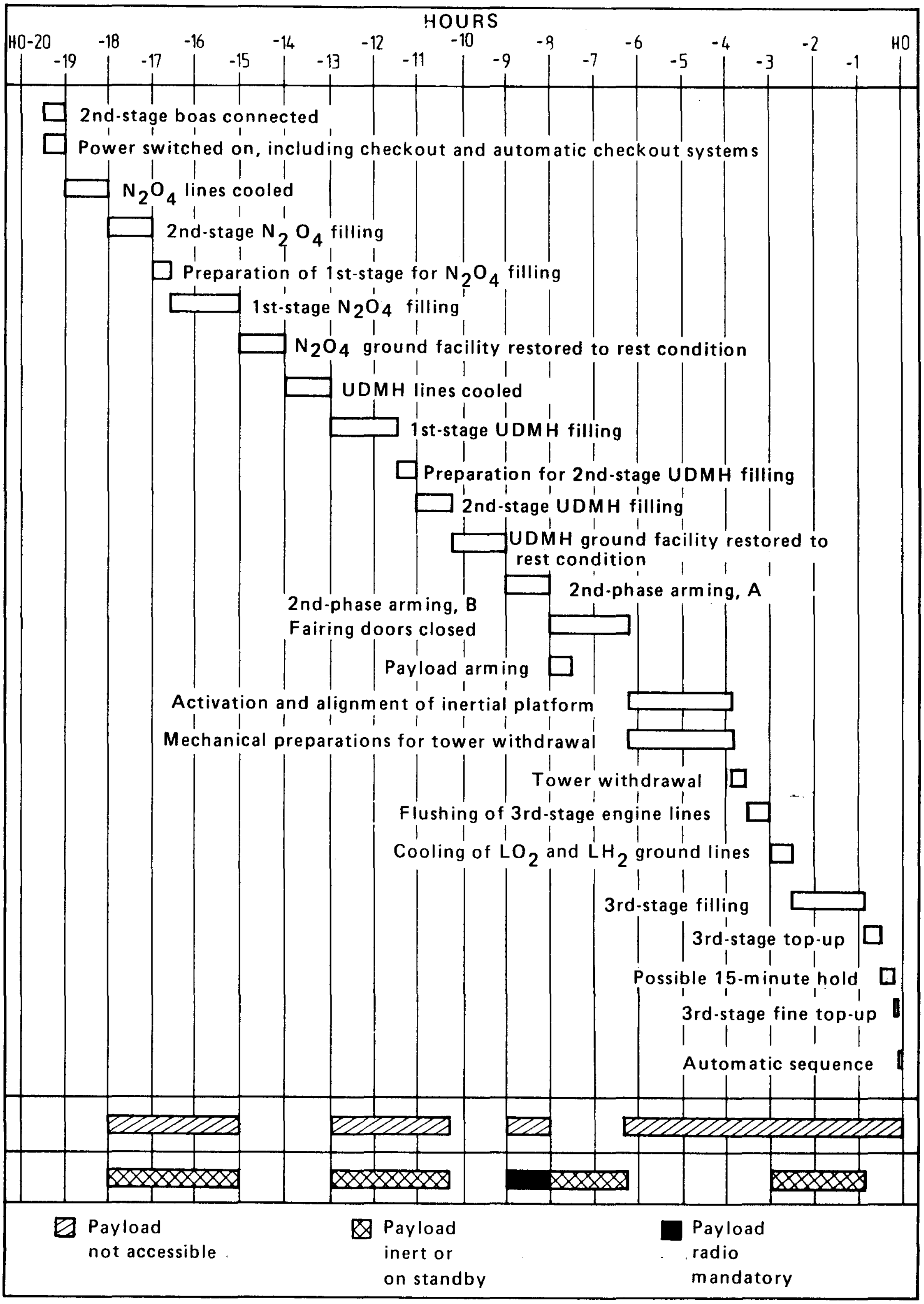


FIG. 4.12 TIME–SCHEDULE HO–20 to HO

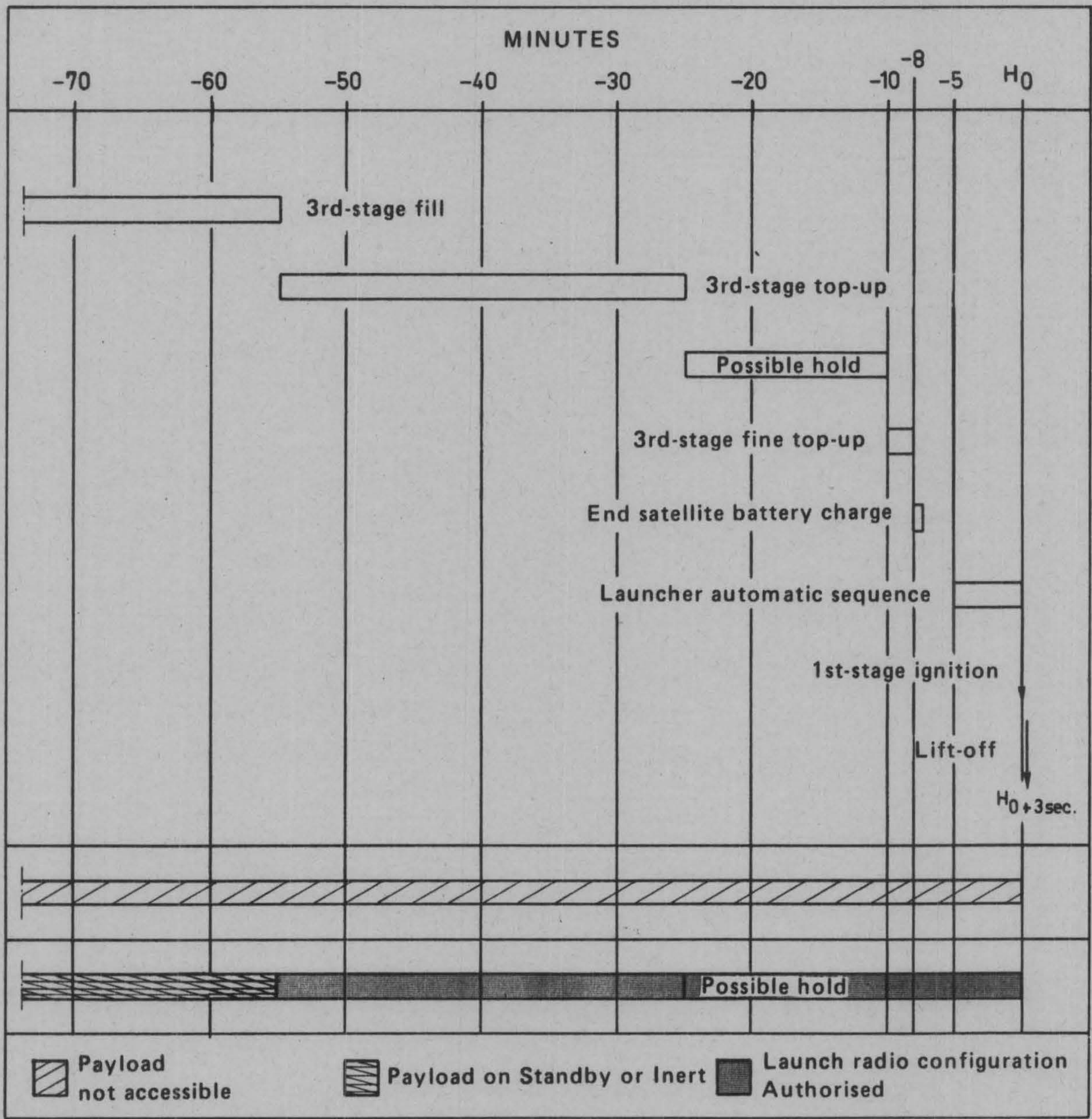
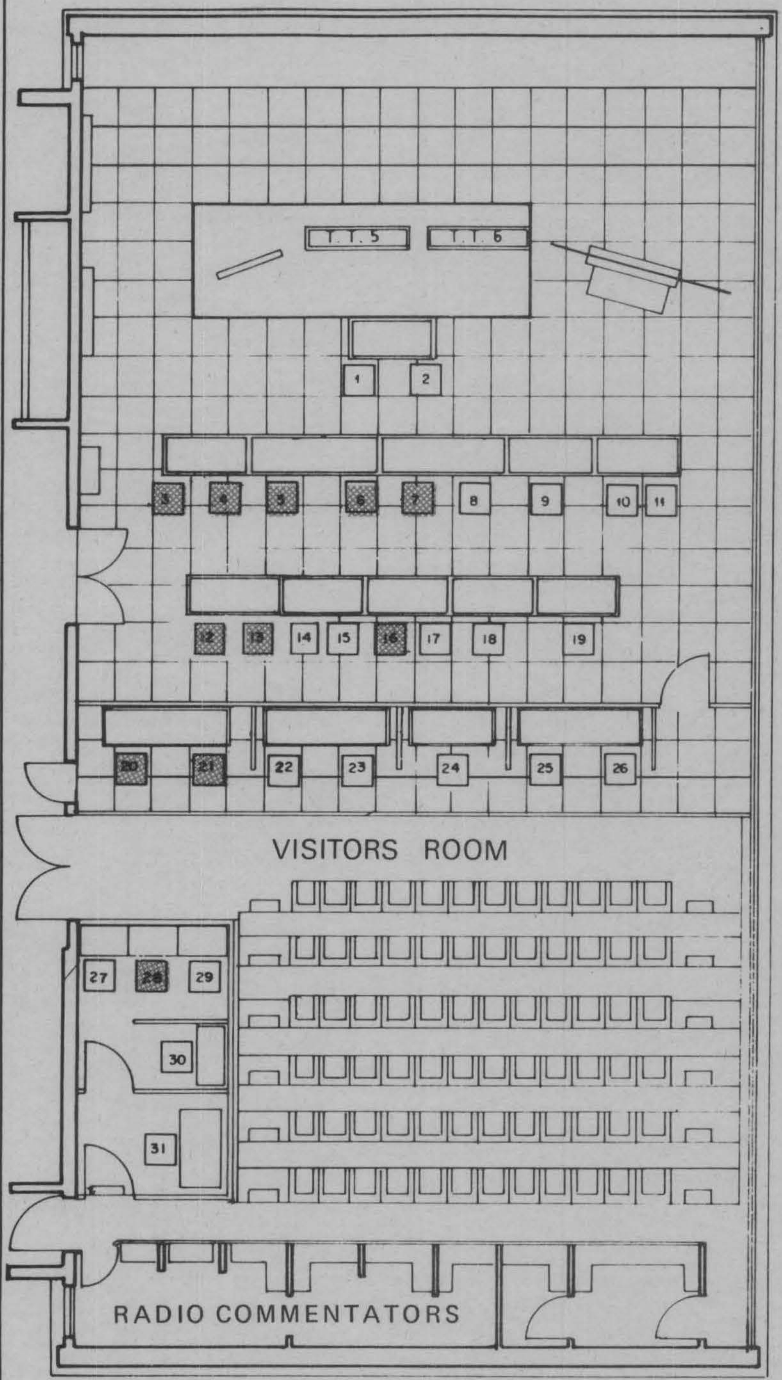


FIG.4.13 COUNTDOWN HO–60 MIN TO HO



1 Plotter
2 Plotter
3 Satellite Project Manager 1
4 Satellite Project Manager 2
5 Assistant Mission Director
6 Mission Director
7 Launch Operations Director
8 Assistant Launch Operations Director
9 Tracking Assistant
10 Telemetry Assistant
11 Communications Assistant
12 Ariane Project Manager
13 Flight Director
14 Head of Organization Division
15 Head of Operations Division
16 Range Director
17 Assistant Director, Technical
18 Head of Instrumentation Division
19 Facilities Support Assistant
20 Payload Network 1
21 Payload Network 2
22 Downrange Station Assistant
23 Ascension Station Operator
24 Salinopolis Station Operator
25 Operations coordination
26 Operations coordination
27 Launcher Technical Commentator
28 Payload Technical Commentator
29 CSG Technical Commentators
30 Public Relations Manager
31 Public Relations Officer

FIG. 4.14 CONTROL CENTRE CONFIGURATION FOR ARIANE LAUNCH

FIG. 4.15 LINKS AVAILABLE TO THE USER DURING LAUNCH
(Will be published in a subsequent issue)

COEL : Launch Site Operations Manager (" Chef des Opérations Ensemble de Lancement ").
Responsible for the preparation, activation and checkout of the launch vehicle and launch-complex facilities. Coordinates all operations on the launch pas (servicing tower and Launch Centre).

RPCU : Satellite Preparation Manager.
(" Responsable de la Préparation Charge Utile ")
Responsible for the preparation, activation and checkout of the payload.

CSEL : Launch Site Safety Officer.
" Chef Sauvegarde Ensemble de Lancement ")
Represents the Safety Officer on the launch site

ROO : Satellite Orbital Operations Manager.
(" Responsable Opérations Orbitales charge utile ").
Responsible for the preparation and execution of the payload orbit flight plan.

ORS : Satellite Ground Station Network Operator
(" Opérateur Réseau Stations sol charge utile")
Responsible for liaison between the CSG Payload Authority and the Payload Orbital Operations Centre.

Control-room configuration for the above personnel and details of communication links available to the user during countdown are indicated in figures 4.14 and 4.15.

4.6. Buildings and associated facilities

4.6.1. Payload preparation complex

(" Ensemble de Préparation Charges Utiles " - EPCU)

This complex is used specifically for payload operations at the CSG. The corresponding facilities are located on a number of sites.

The environmental conditions of the premises are indicated in figure 4.28. A detailed description of the buildings and associated facilities is given in the CSG Manual.

4.6.1.1. Satellite preparation building, S1

This building is located within the Technical Centre. It comprises principally (see figs. 4.16 and 4.17) :

- a clean area of 490 m², equipped with a travelling crane,
  - capacity : 50 kN, covering an area of 490 m², hook height 11 m
- main airlock, length 10 m :
  - access : width 4.5 m
    height 10 m
  - monorail hoist : capacity 100 kN, hook height 11 m

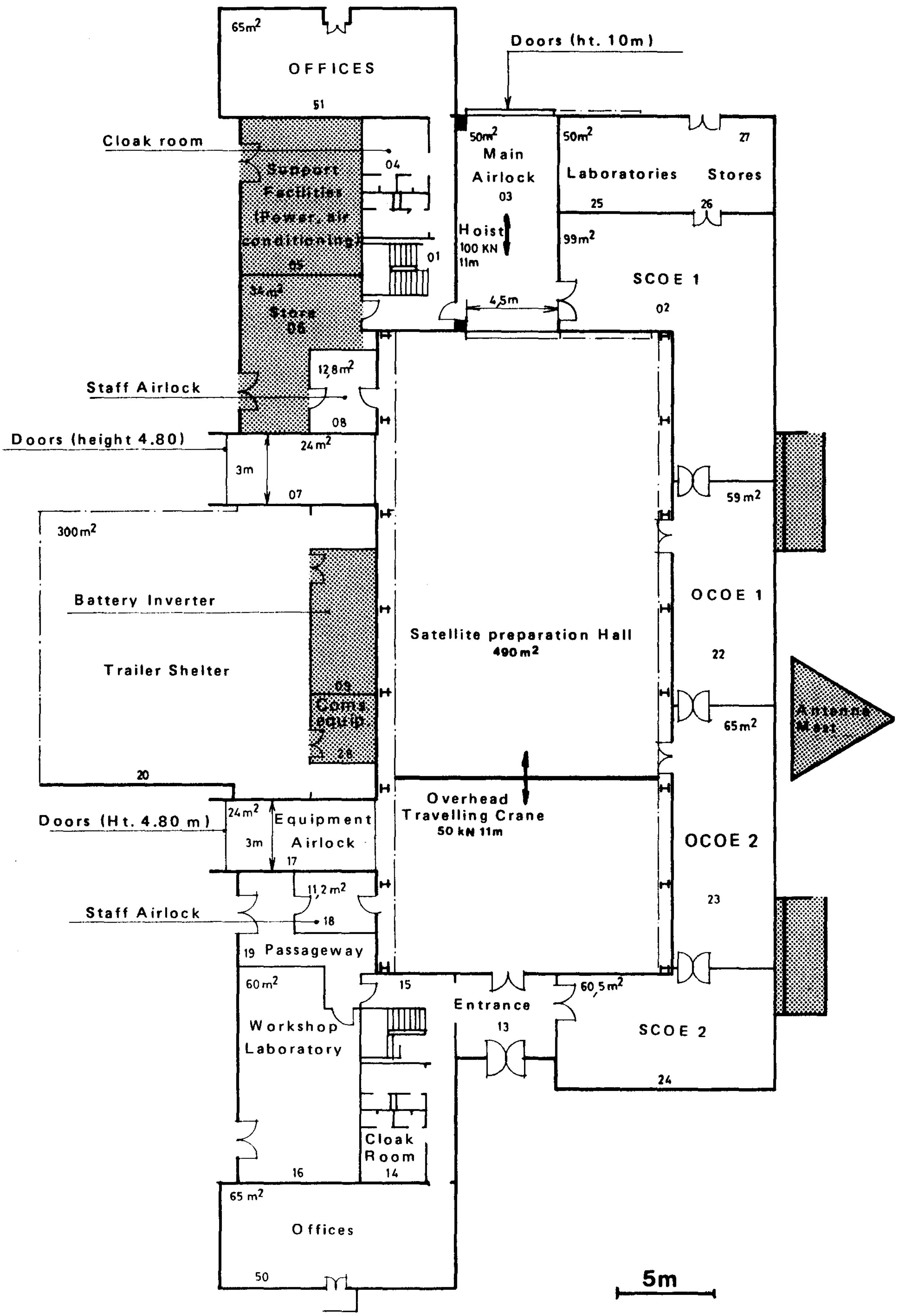


FIG. 4.16 GROUND FLOOR SATELLITE PREPARATION BUILDING (SI)

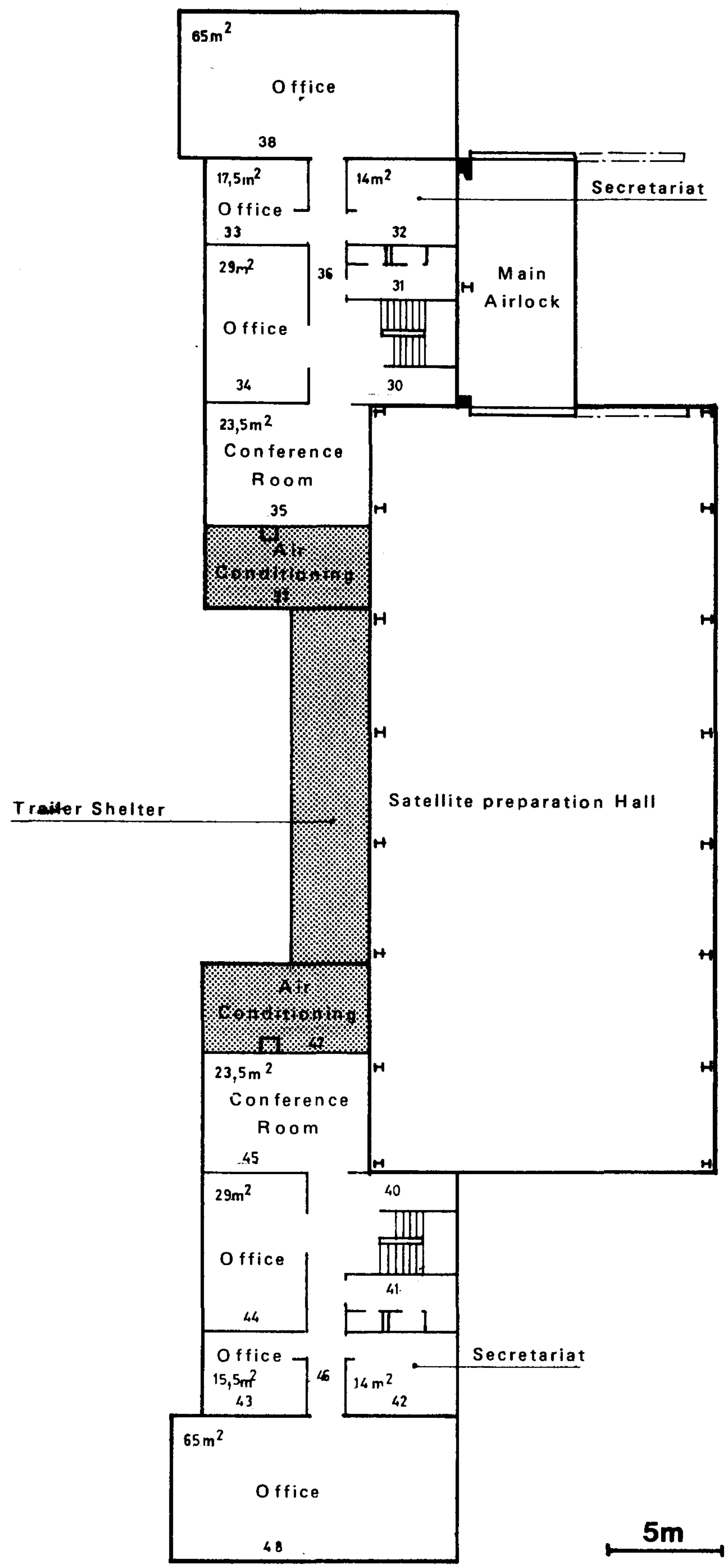


FIG 4.17 SATELLITE PREPARATION BUILDING (SI) – 1st FLOOR

- a secondary airlock, length 6 m :
  - access : width 3 m
    height 4.8 m

- premises with a total area of 285 m² for the satellite checkout equipment, including 124 m² for the OCOE.

- offices and conference rooms
- covered storage and off-loading area of 300 m², not air-conditioned.
- laboratories and store rooms with a surface area of 110 m².
- 42-m mast carrying the telecommunications antennae.
- satellite alignment equipment
- portable equipment for pressurization/depressurization of the hydrazine tanks. This equipment is used in building S1 for detection of possible leaks, and/or in the servicing tower for final pressurization.

- Power supply :
  - clean area : 30 kVA ; 220/380 V three-phase, 50 Hz
    5 kVA ; 120 V 60 Hz
  - checkout systems :
    30 kVA ; 220/380 V three-phase, 50 Hz or
    20 kVa ; 120/208 V, three-phase, 60 Hz
    (no-break supply maintained for 15 minutes)

4.6.1.2. ABM preparation building, S2

This building is located in a protected area of the launch zone to which access is restricted and is specially designed for the preparation of solid-propellant ABMs and pyrotechnic devices prior to integration in the satellite.
Characteristics (see fig. 4.18)

- Environmental conditions : see figure 4.28
- Access under 20-m² canopy
  - door : width 3 m
    height 4 m
  - hoist : hook height 5 m
    capacity 30 kN
- Main Hall
  surface area 97 m²
  travelling crane with capacity 30 kN
  hook height 6 m.
- Secondary Hall
  surface area 48 m²
  monorail hoist, capacity 30 kN
  hook height 6 m

This building includes the following ancillary premises :

- offices
- laboratory with surface area 20 m²
- armoured room, surface area 12 m², for pyrotechnic testing
  Power supply available : 5 kVA ; 220 V, 50 Hz
  5 kVA ; 120 V, 60 Hz (on request).

Facilities are available for testing " MAGE " type motors.

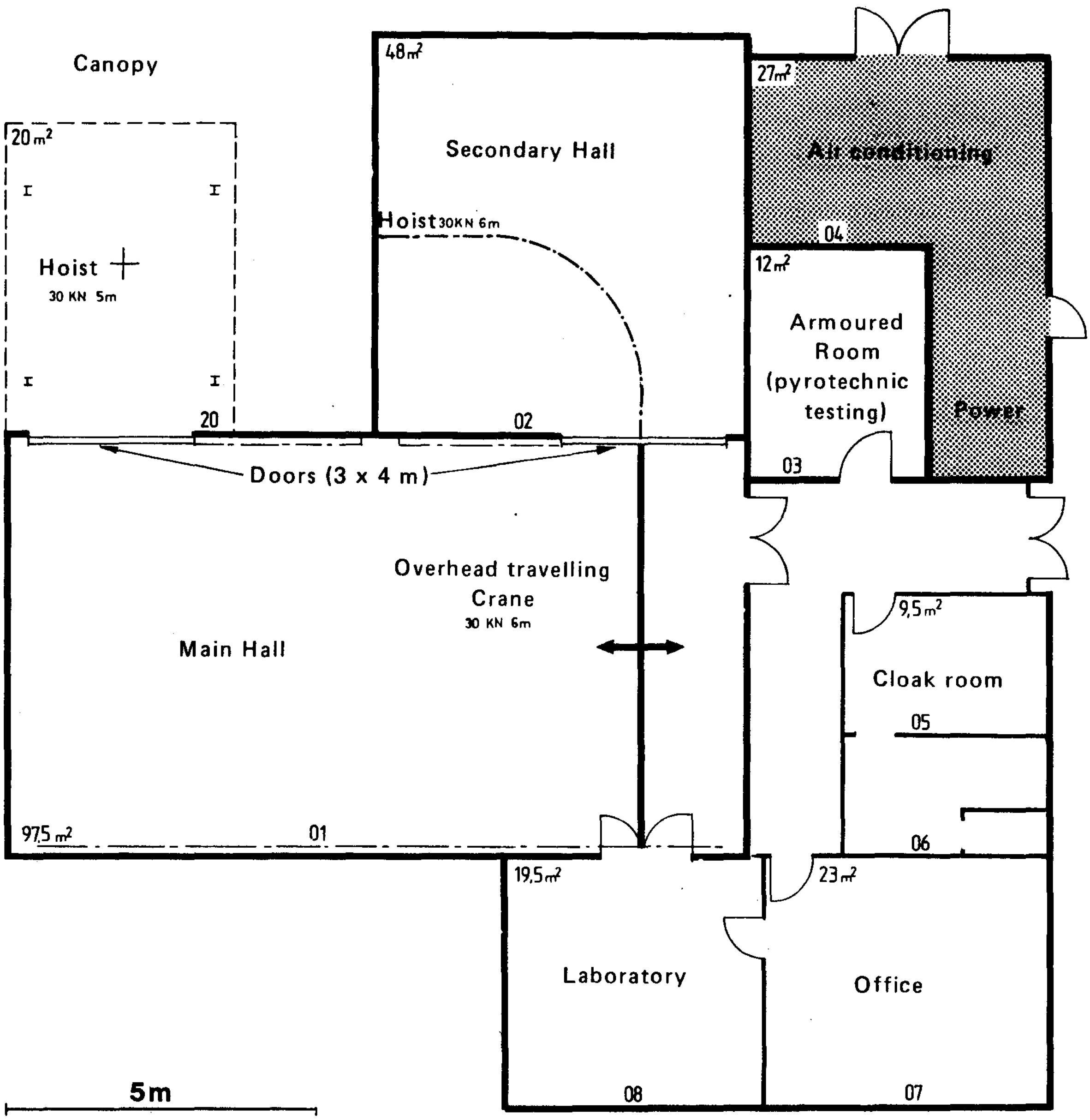


FIG. 4.18 SATELLITE ABM PREPARATION BUILDING (S2)

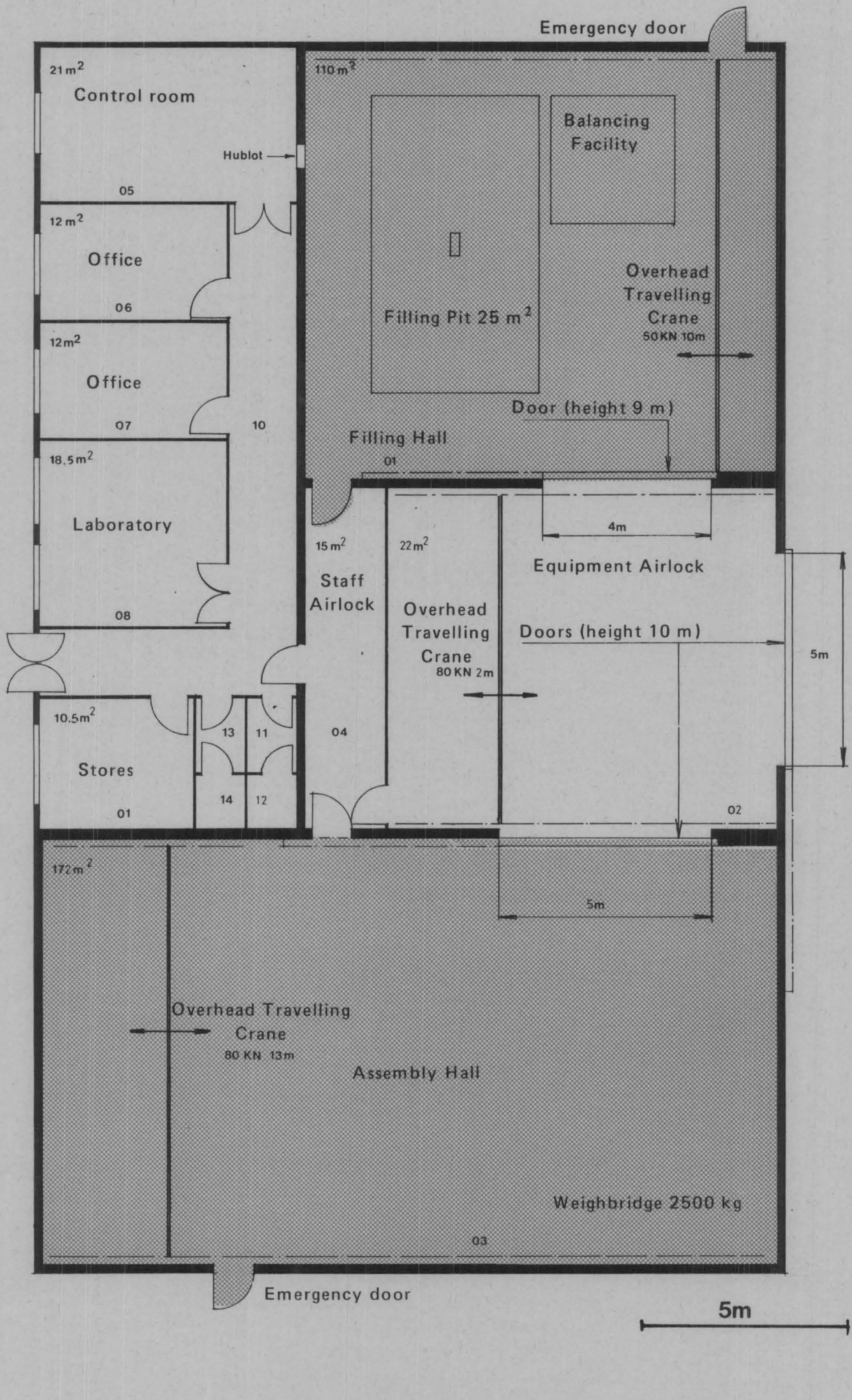


FIG. 4.19 SATELLITE FILLING AND ASSEMBLY BUILDING (S3)

4.6.1.3. Satellite filling and assembly building, S3

This building is located in a protected area of the launch zone to which access is restricted, and is reserved exclusively for payload hazardous operations, such as filling the propellant tanks, assembly of satellite and ABM, and placing the payload in its container (CCU).

Main characteristics (see fig. 4.19).

- Environmental conditions : see figure 4.28
- Filling Hall :
  surface area 110 m², with 25 m² filling zone over grating
  50-kN travelling crane, hook height 10 m
  access via 4 × 9-m door
  balancing machine available
- Assembly Hall :
  surface area 172 m²
  80-kN travelling crane, hook height 13 m
  access via 5 × 10-m door
  weighing machine available
- Access Air Lock :
  surface area 72 m²
  80-kN travelling crane, hook height 12 m
- Armoured test room and workshops
- Associated offices and workshops
- Power supply available : 3 kVA ; 220 V, 50 Hz, for satellite checkout system bays where required (no-break supply)
  5 kVA ; 120 V, 60 Hz.

4.6.1.4. ABM radiography building, S4

This building is located in a protected area of the launch zone to which access is restricted, and is used for the following operations, at user request :

- Cold soaking of the ABM
- X-ray inspection of ABMs with dimensions not exceeding the following :
  - mass : 1.5 tonnes
  - diameter : 1.5 m
  - height 2 m
- X-ray photography and development

This building comprises the following premises (see fig. 4.20) :

- X-ray inspection hall (heavily screened)
  surface area : 75 m²
  30-kN travelling crane, hook height 5 m
  This hall is equipped with an automatic preselection turn-table with 32 positions (11.25° steps)
- Console for remote control of operations
- Cold-soak room
  surface area : 9 m², height : 4 m, for stabilization of the ABM temperature between 0° and 15° C in less than 24 hours.

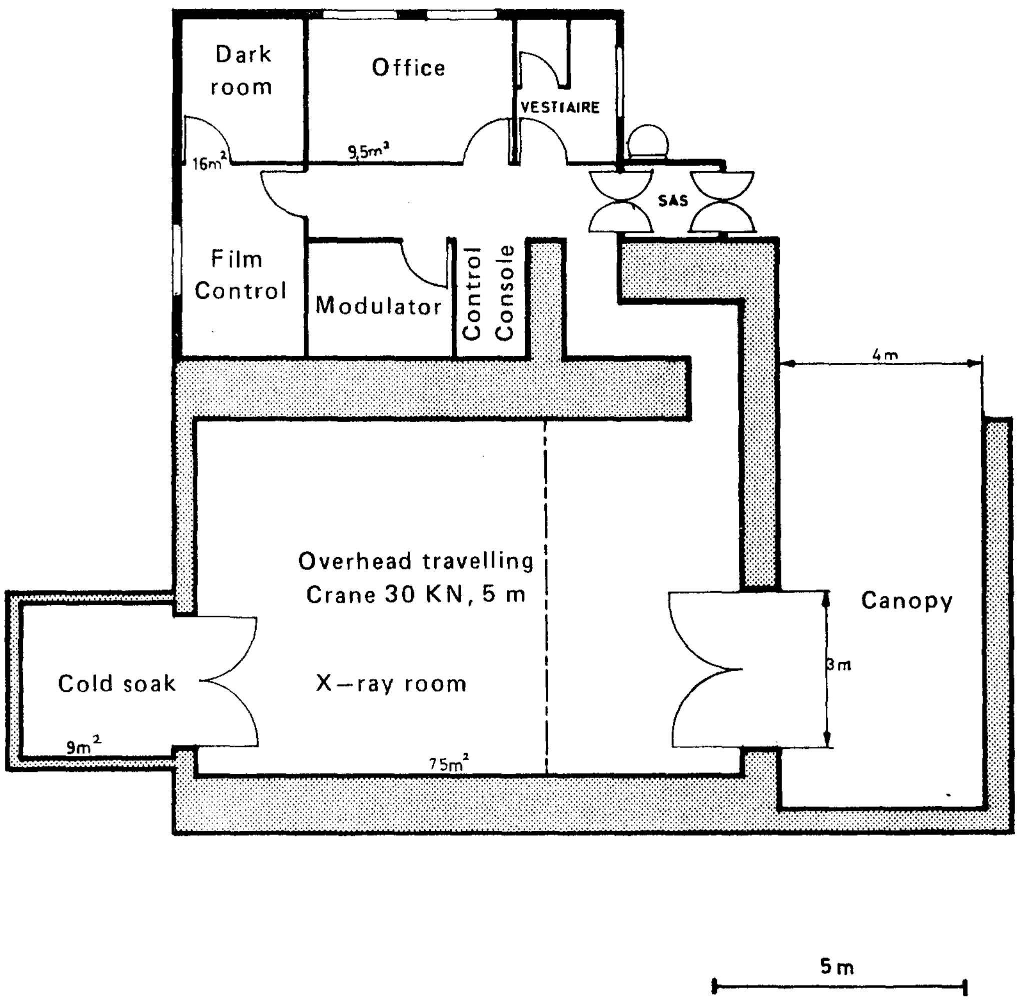


FIG. 4.20 ABM X – RAY BUILDINGS (S4)

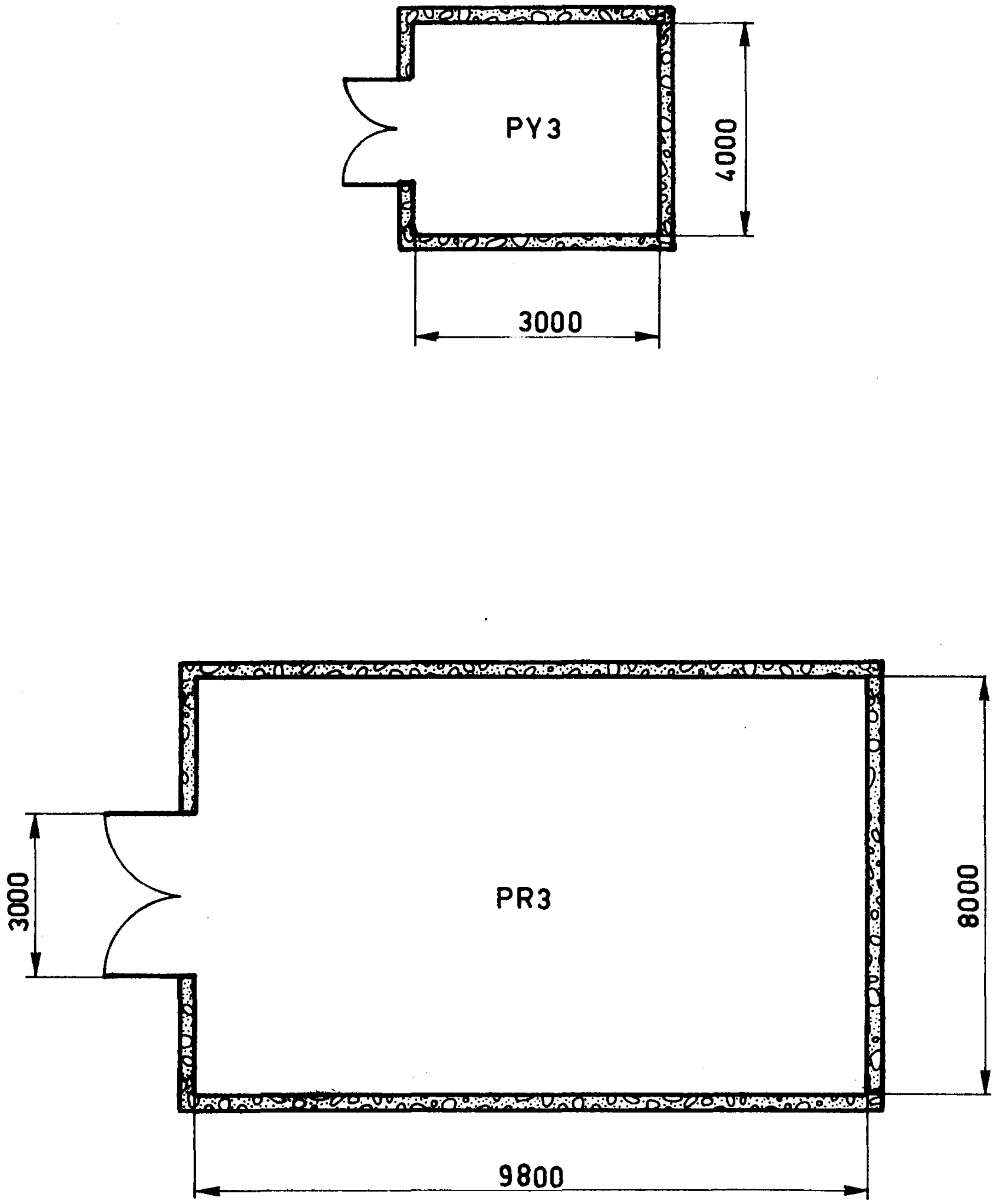


FIG. 4.21 BUILDINGS PY3 AND PR3
SOLID PROPELLANT STORAGE ZONE

- Dark room for development of films, and film examination/analysis room
  surface area 16 m^2
- air-conditioned office and cloakroom, surface area 15 m^2

Environmental conditions : see figure 4.28.

4.6.1.5. Solid-propellant ABM storage, PR 3

Building PR3 is located in the CSG ABM-storage zone.

It is air-conditioned and comprises (see fig. 4.21) a 76-m^2 hall for long-term storage of ABMs in containers, with 50-kN hoist, hook height 5 m.

Access door : width 3 m
height 3.50 m

4.6.1.6. Pyrotechnic-element stores, PY3

This building is located in the CSG ABM-storage zone. It is air-conditioned, and has a surface area of approximately 12 m^2 reserved for storage of electro-pyrotechnic components, prior to their integration in the satellite or ABM motor (see fig. 4.21).

4.6.2. Ariane Launch Zone

This zone includes a number of facilities, two of which are employed by Ariane users (see fig. 4.4.)

4.6.2.1. Ariane servicing tower

This tower is located on the launch-pad area, and is used for erection of the launch vehicle on its launch table. Fully enclosed and air-conditioned, it surrounds the umbilical mast. The upper platform, designated PF8, provides access at payload level :

- from the inside for personnel and small equipment
- from the outside for the payload container (CCU) and fairing.

This platform has a vertical travel of approximately 7 m, and constitutes the floor structure for a clean area, the main characteristics of which are as follows (see figs. 4.22 and 4.23) :

- area approximately 100 m^2, including 25 m^2 airlock with mobile partitions.
- max. hook height 10.34 m
- access width 4.75 m
- travelling crane capacity 2 × 25 kN

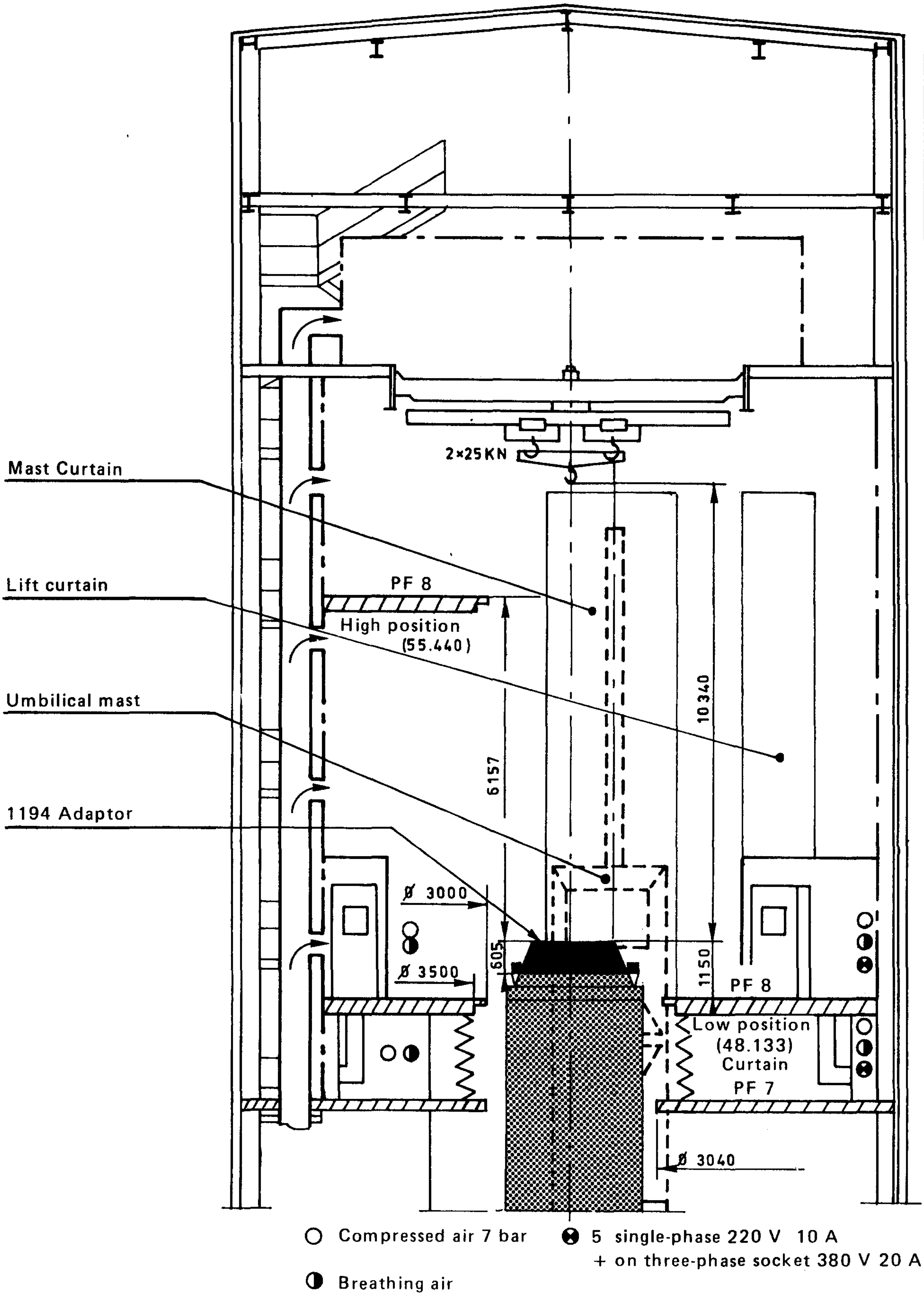
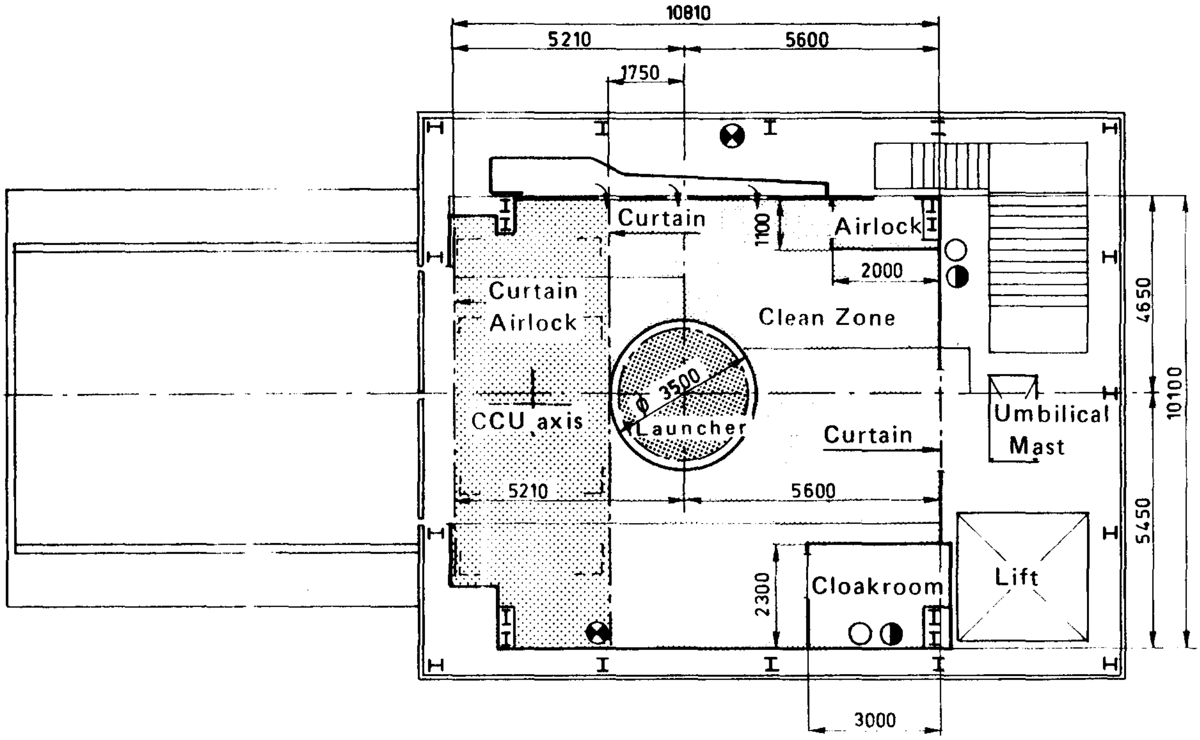


FIG. 4.22 FRONT VIEW OF ARIANE SERVICING TOWER AT PAYLOAD LEVEL



○ Compressed air : 7 bar

◑ Breathing air

⊗ 5 single-phase sockets : 220 V 10 A
+ one 3-phase socket : 380 V 20 A

FIG. 4.23 SERVICING TOWER – OVERHEAD VIEW AT PAYLOAD LEVEL (PF8)

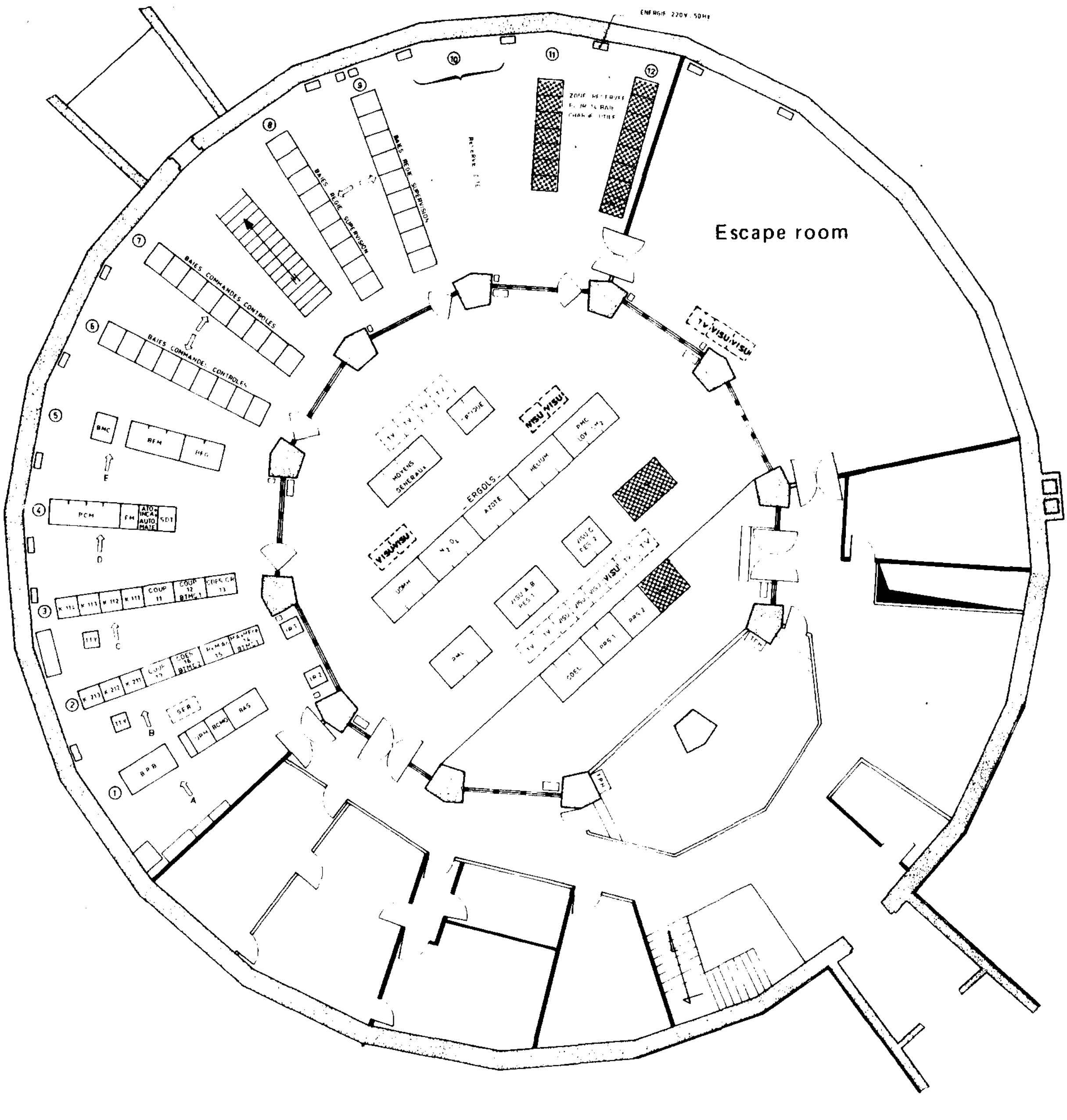


FIG. 4.24 LAUNCH CENTRE

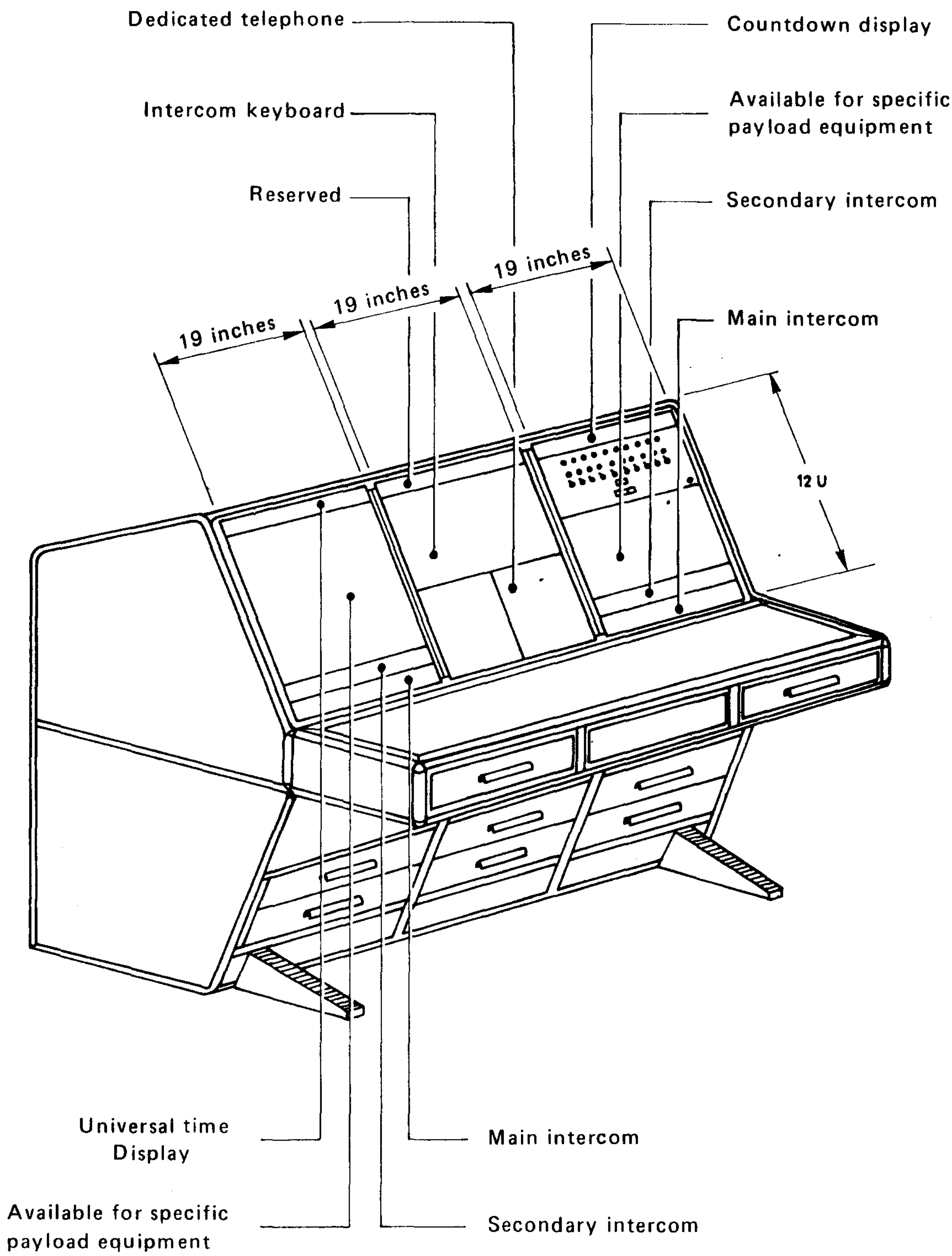


FIG. 4.25 PAYLOAD CONSOLE (PCU)

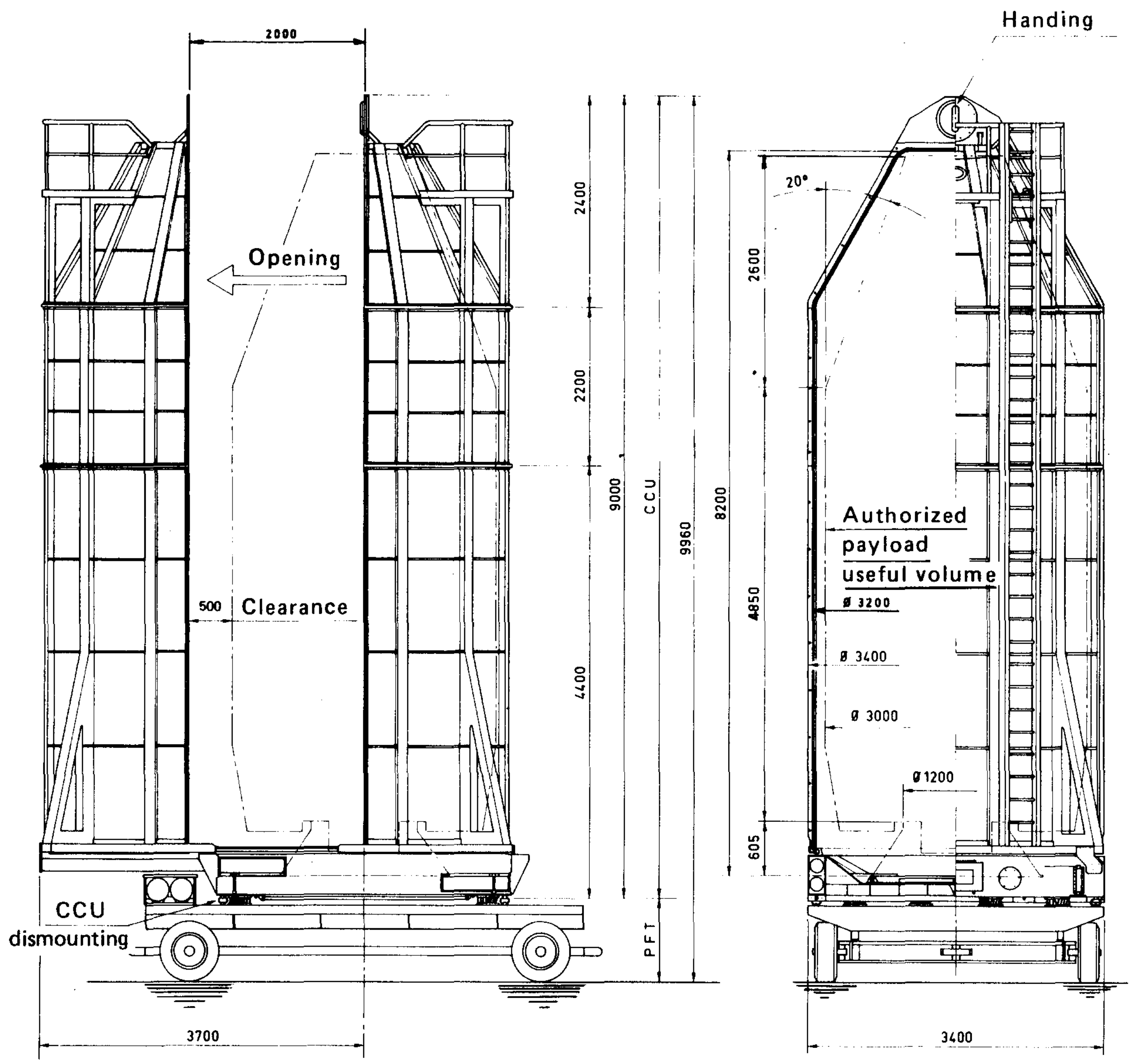


FIG. 4.26 ARIANE PAYLOAD CONTAINER (CCU)

4.6.2.2. Launch Centre (" Centre de Lancement " - CDL)

The CDL is an armoured building, located 300 m from the Ariane servicing tower. It provides protection for personnel and checkout equipment during final preparation and actual launch.

Two areas are reserved for users in this building (see fig. 4.24) :

- An area for the umbilical-link terminal bays and satellite checkout and arming bays, located in the peripheral zone of the CDL. Maximum capacity : 14 bays. Cable connections via the umbilical mast transit through a payload interconnection bay. This area is connected to an antenna base on the CDL roof.
- The Payload Console, located close to the launch console in the central part of the CDL (see fig. 4.25).

4.6.3. Ariane Payload Container (" Container Charge Utile - CCU ")

The CCU is a standard container available to users and capable of transporting the complete payload between buildings S1 and S3 and platform PF 8.

Main characteristics (see fig. 4.26) :

- Two types of payload interface are possible :
  - payload-adaptor type interface (see fig. 3.3a (adaptor 1194), figure 3.9a (adaptor 937) and figure 3.15b (adaptor 1497)).
  - Ariane VEB-type interface (fig. 3.17).
- Continuous dry nitrogen flow during transport.
- Maximum acceleration on payload attachment fittings during transport :
  - vertical : 0.5 g
  - horizontal : 0.3 g

4.6.4. Control Centre

The Launch Control Centre, located in the Technical Centre, houses the senior personnel with authority for the launch vehicle, satellites, networks and the CSG, who formulate the decisions leading up to the launch on the basis of information received from the operational pads. See the CSG Manual for a description of the Control Centre, and para. 4.5. above for details of its utilization.

4.6.5. Storage halls

Various storage areas are available on the CSG. In particular, a building on the sounding-rocket pad can be used for storage of sensitive equipment, and can be air-conditioned if required.

Main characteristics :
- main hall : 14 × 14 m
- access : 4 × m
- 10-kN travelling crane, hook height 5 m.

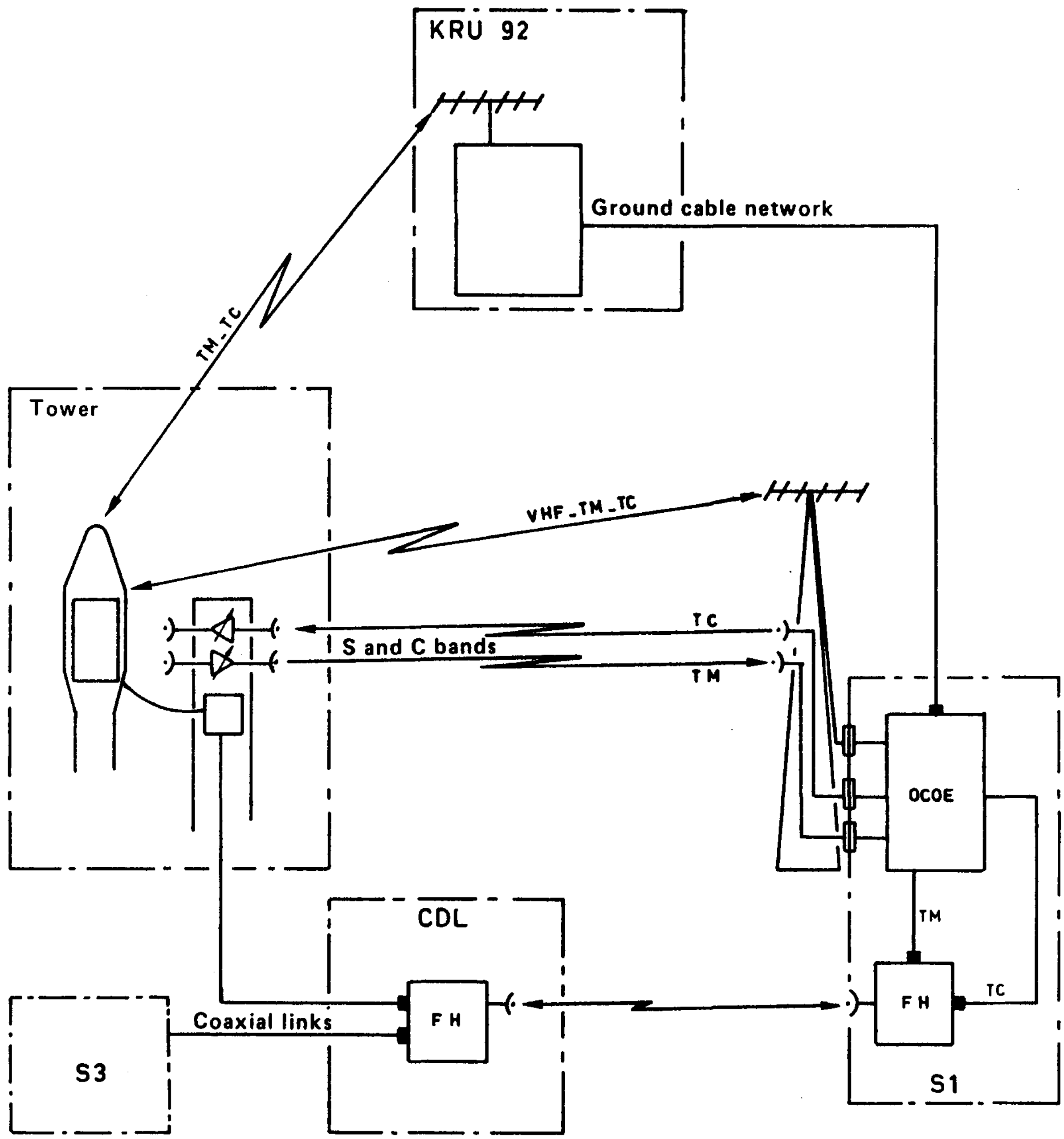


FIG. 4.27 EPCU/TOWER LINKS

BUILDING	CLEANLINESS CLASS	TEMPERATURE (°C)	RELATIVE HUMIDITY (%)
S1 Hall Airlock Checkout equipment Laboratories Offices	100.000 N.A. N.A.	24 ± 1* 24 ± 2* 24 ± 2	55 ± 10 55 ± 10 ≤ 70
S2 Hall Laboratories	N.A	24 ± 2*	≤ 60
S3 Filling hall Assembly hall Access airlock Offices	100.000 N.A	24 ± 1* 24 ± 2	55 ± 10 ≤ 70
S4 X-RAY room Control room Film rooms Offices	N.A	24 ± 2 24 ± 2 24 ± 2 24 ± 2	≤ 60 ≤ 70 ≤ 70 ≤ 70
PR3, PY3	N.A	≤ 28	≤ 60
PF 8 clean area (tower)	100.000	25 ± 2*	50 ± 10
Fairing	100.000	Incoming air : adjustable 15° to 20° C ± 1° C	≤ 15
Launch Centre (CDL)	N.A	24 ± 2	≤ 65
Payload container (CCU)	100.000	24 ± 1. when leaving S 3 (or S1)	≤ 50 with nitogen flushing
Control Centre	N.A	≤ 26	≤ 70
Sounding rocket storage zone	N.A	≤ 28	≤ 70

* Temperature and relative humidity can be mutually adjusted.
Example : T ≤ 24° C, RH = 55 ± 5 %

FIG. 4.28 ENVIRONMENTAL CONDITIONS FOR EPCU AND ASSOCIATED FACILITIES

4.7. Intercommunication

4.7.1. General

In addition to the telephone, telex, intercom and TV systems, the user can also utilize the CSG internal and external data-transmission facilities.

These facilities are either permanent, or established at the user's special request.

4.7.2. Cable

The CSG has a telephone-type cable network, comprising composite cables (pairs and coaxials), for transmission of any signals with telephone characteristics, for pin-to-pin connections, with HF or video transmission on the coaxials, in particular from :

- building S1 to the control centre
- building S1 to the Launch Centre in the launch zone, and from the Launch Centre to buildings S2 and S3.

4.7.3. Radio (see fig. 4.27).

Depending on transmitted power, links can be established between building S1 and a payload :

- inside the servicing tower, with fairing not fitted
- inside the servicing tower, with fairing
- outside the servicing tower with fairing

For this purpose, VHF, S- and C- band antennae are available on a mast at building S1, complete with corresponding feeders.

S- and C- band repeaters are installed on the umbilical mast. The VHF transparency of the servicing tower is adequate without the use of repeaters.

As redundancy for the above links, telemetry and telecommand video systems may be routed over the direct radio-relay link between S1 and the CDL. From the CDL, cable connections extend the links to the satellite, mounted on the launch vehicle (umbilical links). There is also a coaxial cable connecting the CDL to building S3.

Special links can be obtained, in particular between station KRU-92 and the radio-relay station near the Technical Centre (Mont Pariacabo).

4.8. General resources

This paragraph gives a general indication of the facilities and services available to the user at the CSG. For further details, see the CSG Manual.

4.8.1. Technical support

The CSG can make a number of special-purpose workshops available to users, together with corresponding qualified staff. These comprise :

- topographical survey office
- reprography workshop
- propellant analysis laboratory
- carpentry shop
- mechanical and electromechanical workshop
- optical and photographic workshop
- measuring-instrument laboratory

The CSG also provides facilities for the storage of propellants required by the satellites, and corresponding special protective equipment for fuelmen.

Finally, the CSG can make a clean tent (Class 100) with a surface area of 25 m^2 available to the user.

CNES also has special telecommunication facilities, which can be made available to the user :

- international telecommunication and date-transmission network (RESEDA - max. transmission rate 4800 bauds, 16-bit words), for which the operational and computer centre is in Toulouse, France (" Centre Spatial de Toulouse " - CST).
- 1200-baud line to the Darmstadt operations centre, ESOC.

4.8.2. Operational support

The CSG makes the following facilities available to the payload team :

- intercom and telephone network, providing communication between all payload positions,
- internal telex network,
- access to the international telephone and telex networks,
- Closed-circuit TV of current operations in building S3, or on platform PF 8 in the servicing tower, to the control stations (building S1, Launch Centre and control centre).

4.8.3. Logistic support

4.8.3.1. Reception and access

New arrivals are met at Rochambeau Airport and are taken by coach to their respective hotels in Kourou. Persons requiring cars for their mission are taken directly to the technical centre, where they take over their vehicles.

On arrival at the technical centre, all mission personnel are issued with badges, authorizing access to the various technical zones.

4.8.3.2. Accomodation

Hotel reservations are made in connection with the provision of launch services.

4.8.3.3. Vehicles

The CSG makes a certain number of cars available to the payload team, for use within the confines of the CSG. Drivers must hold licences recognized by the French police authorities (international driving permit for example).

4.8.3.4. Shipment and transport

In Guiana, the CSG is responsible for local transport and customs formalities. In order that customs operations may be completed rapidly and smoothly, the CSG must be provided with full information concerning packages dispatched (pro-forma invoice, detailed packing list, etc.).

4.8.3.5. Handling facilities

For local handling of equipment, the CSG can make handling and lifting equipment, with capacities up to 8 tonnes available to the user. Certain facilities not available in the CSG may be rented locally. Capacities range from 2 tonnes to 40 tonnes for lifting gear, and 4 tonnes to 100 tonnes for cranes.

4.9. Launch postponement

The launch slot starts on completion of 3rd-stage filling (H0 −55 minutes), and terminates at the end of the launch window(s) requested by the payload. A launch window has a minimum duration of 20 minutes. The ground facilities are dimensioned, and the launch vehicle designed, so that the daily launch slot, including the various launch windows, may amount to 6 hours.

If the launch does not take place inside the scheduled window(s) during the day's launch slot, the launch will be postponed to the following day. Launch time (H0) is set at the start of the new launch window, and the countdown is restarted at H0 − 6 hours.

At the end of a launch slot during which the launch has not occurred, and assuming the irreversible phase of the automatic sequence has not been reached (H0 −4 seconds), the following operations are carried out :

a) draining and flushing of the 3rd stage,
b) depressurization of the 1st and 2nd stages, down to launch-table waiting pressure.
c) return of the servicing tower to the launch vehicle (approximately 3 hours after the end of the launch slot).
d) launch vehicle, payload and ground facilities placed in countdown-hold configuration.

The payload must not be powered during operations a) and b) above (payload inert).

4.10. Aborted launch

In the event of an aborted launch, namely when the first stage engine ignition command has been sent, but the launch does not take place, a new launch can be envisaged, within a period which obviously depends on the malfunction causing abort on the launch table.

Since the 3rd-stage filling umbilical arms will have been retracted, a much longer emergency draining procedure is followed, as a result of which the tower cannot be brought back on to the launch vehicle until approximately 24 hours after the attempted launch. The payload compartment remains air-conditioned, however, as indicated in para. 3.4.2.4.

4.11. Evaluation and parameters of injection

The Ariane authority provides the user with an evaluation of satellization of the payload between H0 + 30 minutes and H0 + 1 hour. This information includes an estimate of orbit parameters, and 3rd-stage attitude at payload separation.

4.12. Orbital operations support network

A user requiring support from the ESA network for satellite orbital operations can obtain from the Agency a manual giving network station characteristics.

Safety — Chapter 5

5.1. Introduction

The launching of a payload is a complex operation, involving the use of hazardous systems and products. Precautions must therefore be taken to minimize the probability of an accident, and its consequences.

The CSG is responsible for the safety of persons and the protection of property against hazards presented by operations carried out on (and/or controlled from) its territory. CNES has therefore prepared a set of Safety Regulations, which determine the rules applicable at the CSG to all operations involving the use of hazardous systems or products, and the constraints to be observed when defining the launch vehicle, payload and flight plan.

The user must design the payload and its operation in conformity with the CSG Safety Regulations. The following paragraphs are intended to inform the user of the main provisions of these Regulations.

5.2. Submission of hazardous system files

5.2.1. General

The CSG Safety Department is responsible for drawing up the Safety Regulations, and ensuring that they are observed. Any launch from the CSG requires the approval of the Safety Department. This approval covers the following aspects : launch vehicle, payload and flight plan. In order to obtain this approval, the user must demonstrate that his equipment and its utilization comply with the provisions of the Safety Regulations. This demonstration is achieved in a number of stages, by submission of documents defining and describing the hazardous elements and their operation. The submission documents are prepared by the payload authority, and presented to the Ariane authority.

5.2.2. Submission procedure

This procedure aims at a mutual understanding of problems, and their solutions, from the start of the project, in order to avoid loss of time and money resulting from late modifications to the design or manufacture of systems classified as hazardous by the CSG. Documents relating to a given project are submitted in three phases :

Phase 1 :
The payload authority prepares a file containing all documents necessary to inform the CSG of his plans with respect to hazardous systems.

This file contains replies to all the questions in para. 5 (Hazardous Systems) of the Application to Use Ariane, itself contained in para. 6.4. of this Manual.

The CSG studies this file, classifies the hazardous systems submitted, and states any special requirements of the Safety Department.

Phase 2 :
The payload authority presents the hazardous systems manufacturing file, which must meet the requirements stated by the CSG at the end of Phase 1. This file provides information required for producing equipment or facilities at the CSG, to be used during the launch campaign. Finally it states the policy for checking and operating all systems classed as hazardous.

The CSG checks that the manufacturing file complies with the requirements specified in Phase 1, states its intentions concerning the checking of systems classed as hazardous, and indicates the draft procedure to be applied during flight.

Phase 3 :
The payload authority submits a checking and operating procedure for systems classed as hazardous, describing the checking policy and giving details of its execution.

The CSG negotiates such changes as it considers necessary, and accepts the procedure. The latter then becomes the sole authorized document to be applied by the payload authority during the launch campaign, under the control of the Safety Department.

5.2.3. Safety documentation - Time schedule

In the general interest, the safety questionnaire must be completed by the user well ahead of the formal submission phases, in order that the earliest possible allowance be made in the design of the on-board and ground equipment of the payload.

The following time-schedule for formal submissions gives deadlines working backwards from the launch date.

Phase 1

– 28 months	Submission forming part of the Application to Use Ariane (DUA).
– 26 months (or 2 months after submission of the DUA)	Reply to submission, with classification of hazardous systems.

Phase 2

– 25 months	Submission of manufacturing files for systems classified as hazardous.
– 22 months (or 3 months after submission)	Reply to submission.

Phase 3

– 6 months	Submission of hazardous procedures.
– 4 months (or 2 months after submission)	CSG approval.

5.3. Main requirements concerning hazardous systems

5.3.1. Definition

A hazardous system is defined as any assembly comprising a source of danger (chemical products, radioactive elements, pyrotechnics, pressurized systems, etc.) and the equipment used to operate this source (electric, hydraulic or pneumatic circuits, etc.).

A distinction is made between a " Primary Hazardous System ", comprising a source of danger, a source of energy used to operate the system and connecting elements between the two ; and a " Secondary Hazardous System ", which comprises elements connected functionally to a Primary Hazardous System, for the purpose of commanding and monitoring the latter.

5.3.2. Sources of hazard

- Propellants (liquid and solid)
- Chemical products not used for propulsion, but which are :corrosive
  toxic
  flammable
  explosive
- Radioactive products
- Other products with biological effects
- Pressurized fluids
- Radio transmissions
- HT power sources

5.3.3. Constraints on the use of on-board hazardous systems

The presence on board the payload of a source of danger, classified as a hazardous system by the CSG Safety Department, generally makes it necessary for the user to provide safety intercept devices, and to meet the requirements called for in the Safety Regulations for these systems.

These requirements relate to systems that include at least one of the sources of danger listed in para. 5.3.2., the majority of which are associated with electrical, hydraulic or pneumatic circuits used for monitoring and operating such systems. Apart from the general precaution of observing standard practice for the use of hazardous products, the CSG has laid down specific rules for circuits that monitor hazardous systems.

5.3.3.1. Electrical circuits

The electrical circuits connecting the source of hazard and the source of energy required for activating it must be so designed that unwanted operation requires two simultaneous and independent mistakes or two faults not having a common cause. To this end, a two-pole intercept system for interrupting the circuits is used ; one intercept is located close to the source of hazard and the other close to the energy source. These intercepts, which do not absolve the user from providing normal arrangements for operating the hazardous source, are under the control of the CSG Safety Officer.

The intercept close to the source of hazard can be under local manual control (set of plugs or sockets of a special type and form). The intercept close to the source of energy must be telecommanded, and the activation of the circuit requires consecutive action by two separate operators. In the released condition, the intercept control system interrupts the hazardous circuit. Figure 5.1. illustrates the principle of cable intercepts for a hazardous circuit, and the associated control and monitoring circuits.

For further details, see para. 4.1.2.1. of the CSG Safety Regulations.

In cases where the user provides for operation of a hazardous circuit via the telecommand and telemetry systems of the payload, monitoring data must be displayed on the Payload Safety Console. Special procedures can be envisaged for telecommanded or computer-controlled operations.

A cable intercept on circuits operating hazardous systems, the degree of risk of which is classified as " catastrophic " (see Safety Regulations, chapter 1, para. 1.1.2.), is mandatory.

In wiring hazardous electrical circuits, all necessary technical precautions must be taken to ensure the highest possible degree of independence and insensitivity with respect to other circuits, hazardous or not, and whether electrical, radio or electromagnetic.

The following practical arrangements must be adhered to :

- Power sources (batteries) will be dedicated. As far as possible, all batteries will be of the sealed type.
- High-reliability components will be used in electrical circuits, and will be suitable for the environmental conditions concerned.
- Electrical controlled igniters and detonators will be so designed as to be highly insensitive to external interference. Their main characteristics must be as follows :
- RF sensitivity :
  When the equipment is in a field of 2 W/m^2 (for frequencies 0.15 - 50 MHz) or 100 W/m^2 (for frequencies over 50 MHz), the induced power must be at least 20 dB below the RF sensitivity threshold (if known) or 20 dB below the highest direct current not operating the devices. If these values are not achieved, an RF filter must be fitted so as to compensate for the circuit's inadequacy.The filter-igniter assembly must be screened.

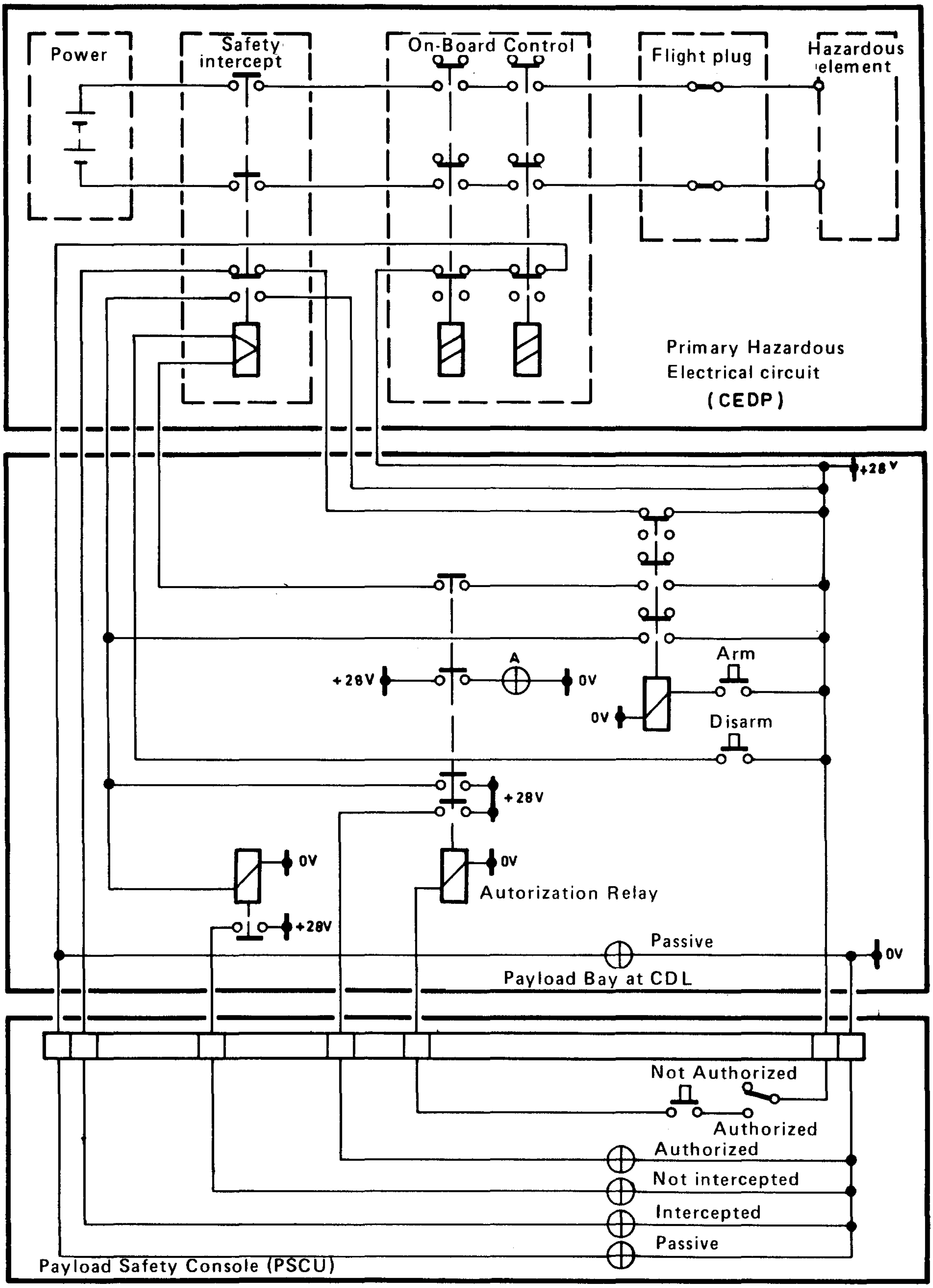


FIG. 5.1 SAFETY INTERCEPT PRINCIPLE DIAGRAM

- Sensivity to electrostatic discharges :
  The equipment must be able to withstand, withoud functioning and without degradation, a discharge of :
  - 25 000 V supplied by a 500 pF capacitor through a 5000 Ω non-inductive resistance ;
  - 10 000 V supplied by a 500 pf capacitor without a resistance.

  In the first case, the voltage should be applied to the terminals of the equipment, and in the second between the short-circuited terminals and the equipment chassis.

  Waivers up to 5000 V may be granted, but in such cases the firing circuit must allow electrostatic charges to leak away continuously.

- If the electrical circuits of the payload and/or the associated ground hardware have to be supplied with power in specific environmental conditions (filling of the launch-vehicle stages or the payload with propellant), they must be designed so as to provide a form of protection suited to the environmental mode :
  - anti-deflagration
  - overpressurization
  - intrinsic safety
  - etc.

5.3.3.2. Inhibition or destruction in flight

The Ariane launch vehicle is equipped with devices for destruction in flight, initiated either automatically or by telecommand, in order to minimize the likelihood of endangering protected areas in the event of malfunction of one of the stages. To this same end, if the payload comprises major sources of hazard, the CSG may require that it be equipped with systems designed to :

- guarantee that its probability of impact in a protected area is negligible in the event of telecommanded or automatic destruction of the launch vehicle ;
- minimize the likelihood of endangering a non-protected area.

5.3.3.3. Hazardous fluid systems

Pressurized systems may constitute a hazard because of the energy released following a burst.

Protection against such hazards is mainly afforded by the value of the safety factor (ratio between rupture pressure and operating pressure) applied to the system.

5.3.3.3.1. On-board pressurized fluid systems

The following rules must be applied :

- pressure vessels must have a safety factor of at least 2.
- associated equipment (pipework, valves, etc.) must have a safety factor of at least 4.

Before operation at the CSG, pressurized fluid systems must be subjected to a test pressure of at least 1.5 times operating pressure. Proof must be provided (certification) by the user that this test has been carried out successfully at a date that guarantees that the test remains valid during operations at the CSG. The above qualification conditions being met, the user must comply with the following conditions, irrespective of location at the CSG :

- maximum operation pressure must not be exceeded at any time.
- authorized personnel may only have access to pressure vessels and associated equipment if the safety limit or factors accepted by the CSG Safety Department are observed.
- tests must be executed in accordance with a written procedure, approved by the CSG.

Any work (e.g. repairs) on a pressurized system will only be undertaken after its depressurization to a safety factor of at least 4.

5.3.3.3.2. Pressurized fluid systems on the ground

Ground fluid systems must comply with French industrial safety standards. The CSG Safety Department will advise the user which standards must be observed, at the time of Phase 1 submission.

5.3.3.3.3. Fluid systems involving hazardous products

In designing such systems, account must be taken of the hazardous properties of the fluids (corrosion, toxicity, etc.), and of the regulations applying to pressurized circuits.

Where a system carries a cryogenic fluid, precautions appropriate to the properties of the substance must be taken, and the constraints on material strengths observed. In particular, these precautions apply to any cryogenic circuit the temperature of which is below 90.3° K (liquefaction temperature of oxygen at atmospheric pressure).

5.3.3.4. Radioactive systems

The CSG has not laid down any specific rules for these systems. A corresponding submission will be presented for joint examination of the problem, from the standpoint of French industrial safety regulations (legislation covering the protection of workers against ionizing radiation).

Special precautions are only required in the following cases :

- radioactive substances with total activity exceeding 0.1 microcurie.
- radioactive substances the mass activity of which exceeds 10 microcurie/kg.
- equipment emitting ionizing radiation, if the radioactive materials included generate a dose equivalent rate exceeding 0.1 millirem/hour at a distance of 0.1 m from the equipment.

5.4. Limits of responsibility

The payload authority is responsible for the design and operation of all payload systems classified as hazardous and their associated ground equipment.

These sources of danger may be handled by CSG personnel, but under the responsibility of the payload authority.

If the handling of these sources requires the use of specific equipment supplied by the user, such equipment must comply fully with the requirements of the CSG Safety Regulations.

5.4.1. Safety of payload team

This comes under the general heading of the safety of personal working on the CSG, governed by the CSG Safety Regulations.

Any activity involving a source of danger is reported to the CSG, which takes all necessary steps to provide and operate adequate collective protection equipment, and to alert the emergency facilities in the event of an accident. Each member of the payload team must comply with the safety rules, with respect to his own personal protection. This is checked by the CSG, which issues corresponding work authorization.

On special request from the user, the CSG can provide specific protection for members of the payload team.

On arrival at the CSG, payload personnel will be given a course of instruction covering the safety rules to be observed and designed to familiarize them with the operation of certain facilities.

5.4.2. Payload activities on the Ariane launch site

All payload activities on the launch site are executed in accordance with instructions given in the corresponding procedures prepared by the user, and by the Ariane authority where combined operations are concerned. These procedures are approved by the CSG Safety Department, and are covered by formal authorization issued by the CSG.

5.4.3. Constraints

Beyond a certain phase in the launch preparation, the launch vehicle and the payload represent hazards for each another because of the hazardous sources that they contain.

The restrictions on the operation of and access to the payload are directly linked to the launch countdown. See paragraph 4.2.2.

Restrictions on checking the payload, such as radio silence, may be imposed during periods of the countdown.

Documentation Chapter 6

This chapter describes the documentation system that comes into force as soon as the user selects Ariane as his launch vehicle. Figure 6.1. gives a list of documents and other information to be exchanged. Figure 6.2. gives the time-schedule for preparation and utilization of all these documents.

6.1. Documentation system

The documentation system involves the use of the documents listed below, prepared by the user or the Ariane authority.

6.1.1. Application to use Ariane

(" Demande d'utilisation Ariane " - DUA) (Document n° 1)

The user prepares an application to use Ariane, in order to define the satellite's requirements with regard to : mission and trajectory ; dynamic, thermal and radio environment ; accessibility and radio-transparency ; ground and umbilical cables ; fluids used ; requirements at the CSG, etc. This application also contains a brief satellite development plan, and scheduled tests. It gives a description of the satellite, and answers the Safety Questionnaire with respect to hazardous systems. The DUA thus covers the Phase-1 safety submission (see chapter 5).

The DUA is self-sufficient as a document, and has no recourse to other documents.

This document is designed to officialize contacts between the Ariane and payload authorities, for the purpose of preparing the launch system/payload interface control file (" Dossier de Contrôle d'Interface " - DCI). Updating of the DUA is not necessary after distribution of the DCI.

A format for the DUA is given in para. 6.4.

6.1.2. Launch system/payload interface control file

("Dossier de contrôle d'interface système de lancement / charge utile - DCI ") (Document n° 2).

This is the basic document which is binding on the Ariane authority and the user with respect to technical aspects and operational facilities.

The Ariane authority prepares the DCI in response to the DUA. The DCI collates all interface requirements common to the launch system and satellite, and demonstrates their respective compatibility. It sets out an interface-modification management procedure, and is updated as the project progresses. On approval of the DCI by the payload authority and Ariane authority, this procedure comes into force.

The user is required to indicate his main requirements concerning satellite preparation operations at the CSG at the latest 14 months before launch. Beyond this deadline, only minor additional requests can be introduced.

The attention of the user is drawn to the fact that basic launch-vehicle flight-mission data are frozen 8 months ahead of launch. These data correspond to those contained in the DCI at this stage.

A format for the DCI is given in para. 6.5.

6.1.3. Mission analysis file

(" Dossier d'analyse de mission " - DAM) (Document n° 3)

The mission analysis undertaken by the Ariane authority covers the following :

a) study of orbit-injection trajectory, separation phase and corresponding dispersions ;
b) analysis of dynamic environment (coupled analysis) ;
c) analysis of thermal environment ;
d) study of pogo stability and control during 3rd-stage flight ;
e) study of radio compatibility.

The following information concerning the payload is required from users :

a) mass, alignment and inertias (extracted from the DCI) ;
b) finite-element matrix-type three-dimensional structural model of the satellite (for coupled, pogo and control analyses) ;
c) thermal model ;
d) radio characteristics (extracted from the DCI).

Analysis of the mission is confirmed in a file (DAM), which is used to update values given in the DCI (concerning : trajectory ; flight sequence ; disengagement margins on separation ; short-term collision risks ; orbital and attitude accuracy on separation ; performance ; dynamic, radio and thermal environment, etc.).

6.1.4. Satellite environment test file (Document n° 4)

Initially, the user is required to provide the Ariane authority with the satellite environment test plan, describing the test called for in paragraphs 3.4.1.5. and 3.5.4. This plan is analysed at the time of the Mission Analysis Review (see para. 6.2.1.).

The user will then submit analysis and synthesis files resulting from the tests. These files are analysed at the Satellite Flight Readiness Review (" Revue d'Aptitude au Vol Satellite " RAVS) (see para. 6.2.3.).

6.1.5. Safety submissions

These are described in chapter 5.

It should be noted that the user is not required to prepare a new document to meet the safety submission requirements, but has only to submit the files normally accompanying satellite development (specifications for Phase-1 submission, definition files for Phase 2, and procedures for Phase 3).

6.1.6. Application to use ESA network (Document n° 5)

In this application, the user indicates the telemetry, telecommand and tracking facilities in the ESA network stations that he wishes to use at the time of the launch and when the satellite is in transfer orbit.

6.1.7. Launch application

(" Demande de lancement " - DL) (Document n° 6)

The launch application is prepared by the Mission Head (CM) and his Assistant (ACM) (see para. 4.5.), and is a synthesis covering both launch-vehicle and payload aspects.

It takes account of all requirements relating to the payloads as defined in the DCI.

It defines the mission objective, the launch characteristics, the general organization, the time-schedule, and the assistance required in the form of personnel and facilities. It is addressed to the CSG and the other departments concerned in the launch. The launch application is signed by the CM and ACM.

The DL is submitted to the user for comment. It is updated systematically up to establishment of the Launch Order (see para. 6.1.10).

6.1.8. Satellite operations plan at CSG

(" Plan des Opérations satellite " - POS) (Document N°7)

The user prepares a satellite operations plan at the CSG, defining operations executed on the satellite on its arrival in Guiana : transport, integration and checkout before assembly, and operations on the Ariane launch-pad area. The POS defines the arrangements for these operations, and describes the facilities required for their execution.

Eight months before launch, the Ariane authority sends its comments to the user, to indicate any modifications required to the POS.

The format for this document is shown in para. 6.6.

6.1.9. Combined operations plan
(" Plan des opérations combinées " - POC) (Document n° 8)

Prepared by the CM and ACM from the POS, insofar as the satellite is concerned, this document details the technical characteristics of the launch corresponding to the requirements imposed by the payload and launch vehicle. It defines the substance of the tasks to be executed, and their breakdown and sequence, from the arrival of the launch vehicle and satellite in Guiana up to processing and evaluation of the actual launch data.

The POC is sent to the user for comment. The user will be expected to operate at the CSG on the basis of a fully compatible POS.

6.1.10. Launch Order
(" Ordre de lancement " - OL) (Document n° 9)

The launch order is prepared by the CSG Operations Director, in response to the launch application. It lays down the following in detail : organization, facilities employed, services provided, time-schedule of operations, and countdown leading to actual launch. As from 4 months before launch, the launch order becomes the reference document for all operations at the CSG, and in particular for the countdown.

The launch order is updated systematically, immediately on its approval by the CSG and CM.

6.1.11. Satellite operation procedures at CSG (Document n° 10)

These procedures are prepared by the user for each operation defined in the satellite operations plan (POS). All procedures covering the operation of systems classified as hazardous and those concerning personnel safety are submitted to the CSG Safety Department for approval (see chapter 5).

6.1.12. Combined launch vehicle/payload operation procedures
(Document n° 11)

A distinction is made between 2 types of combined operation : those requiring procedures specific to each authority, and those requiring common procedures. Common procedures are prepared by the Ariane authority, and submitted to the payload Authority for approval.

6.1.13. Payload mass characteristics

The mass of the payload in its final launch configuration must be notified to the Ariane authority prior to the Launch Readiness Review (RAL) (see para. 6.2.4.).

6.1.14. Injection data

The Ariane authority can transmit data to the user, in quasi-real time (before H0 + 60 min.), concerning the position, velocity and attitude of the payload at the instant of launch vehicle/payload separation.

6.1.15. Orbital tracking operation report (Document n° 12)

The Ariane authority requires the user to supply it with orbital tracking data for the first few satellite orbits, to enable redundant evaluation of launch-vehicle performance.

6.1.16. Launch evaluation report (Document n° 13)

On the basis of processed launch-vehicle telemetry data and tracking data, the Ariane authority draws up a report on the launch operations, indicating the performance obtained and checking the behaviour of the launch vehicle and its subsystems.

That part of this report covering all aspects of launch vehicle/payload interfaces is communicated to the user.

6.2. Launch vehicle/payload reviews

6.2.1. Mission analysis review
(" Revue d'analyse de mission " - RAM)

Apart from aspects specific to the launch vehicle, one of the objectives of this review, held by the Ariane authority, is to check the compatibility of payload environmental test plans in the light of mission analysis results, and to confirm the main mission parameters. The conclusions of the RAM lead to updating of the DCI.

The payload authority is invited to attend the RAM.

6.2.2. Launch-vehicle flight readiness review
(" Revue d'aptitude au vol du lanceur " - RAVL)

The purpose of this review is to check that the launch vehicle, following acceptance tests in Europe, is technically capable of executing its mission. One activity covered by the RAVL concerns examination of the launch-vehicle/payload interfaces, with particular reference to the DCI.

This review is held by the Ariane authority, and shipment of the launch vehicle to the CSG is contingent on it.

The user is invited to attend the RAVL.

6.2.3. Satellite flight readiness review
(" Revue d'aptitude au vol du satellite " - RAVS)

NO	DOCUMENT	CODE	AUTHORITY	CHARACTER
1.	Application to use ARIANE	DUA	User	
2.	Launch system/payload interface control file	DCI	ARIANE	A-C
3.	3.1. Finite-element matrix-type three dimensional satellite model		User	
	3.2. Satellite thermal model		User	
	3.3. Mission analysis file	DAM	ARIANE	
4.	Satellite test files			
	4.1. Environmental test plan		User	
	4.2. Results of environmental tests		User	
	4.3. Launch vehicle/satellite radio compatibility test plan		ARIANE	O
	Safety submissions		User	
5.	Application to use ESA network		User	O
6.	Launch application	DL	ARIANE	C
7.	Satellite operations plan at CSG	POS	User	
8.	Combined operations plan	POC	ARIANE	
9.	Launch order	OL	ARIANE	A-C
10.	Satellite operations procedures at CSG		User	A
11.	Combined launch-vehicle/payload operations procedures		ARIANE	A
	Payload mass characteristics		User	
	Injection data		ARIANE	
12.	Orbital tracking evaluation report		User	
13.	Launch evaluation report		ARIANE	

A : Approved

C : Configured

O : Optional

FIG. 6.1 ARIANE/PAYLOAD DOCUMENTATION

The Ariane authority requires to be represented at this review, normally held by the user before shipment of the satellite to the CSG. In particular, the Ariane authority uses this opportunity to obtain the results of environmental acceptance tests, and the latest real inertial and mass characteristics of the satellite.

6.2.4. Launch readiness review

(" Revue d'aptitude au lancement " - RAL)

This review is held at the CSG after assembly of the satellite on the launch vehicle, and closure of the fairing. Its purpose is to authorize final operations leading up to launch. It is held by the CM, and takes account of all checks made of the launch vehicle, payload and launch facilities.

6.3. Documentation time-schedule

The time-schedule for the issue of the documentation and the holding of reviews is given in figure 6.2.

By convention, all dates are considered as " at latest ".

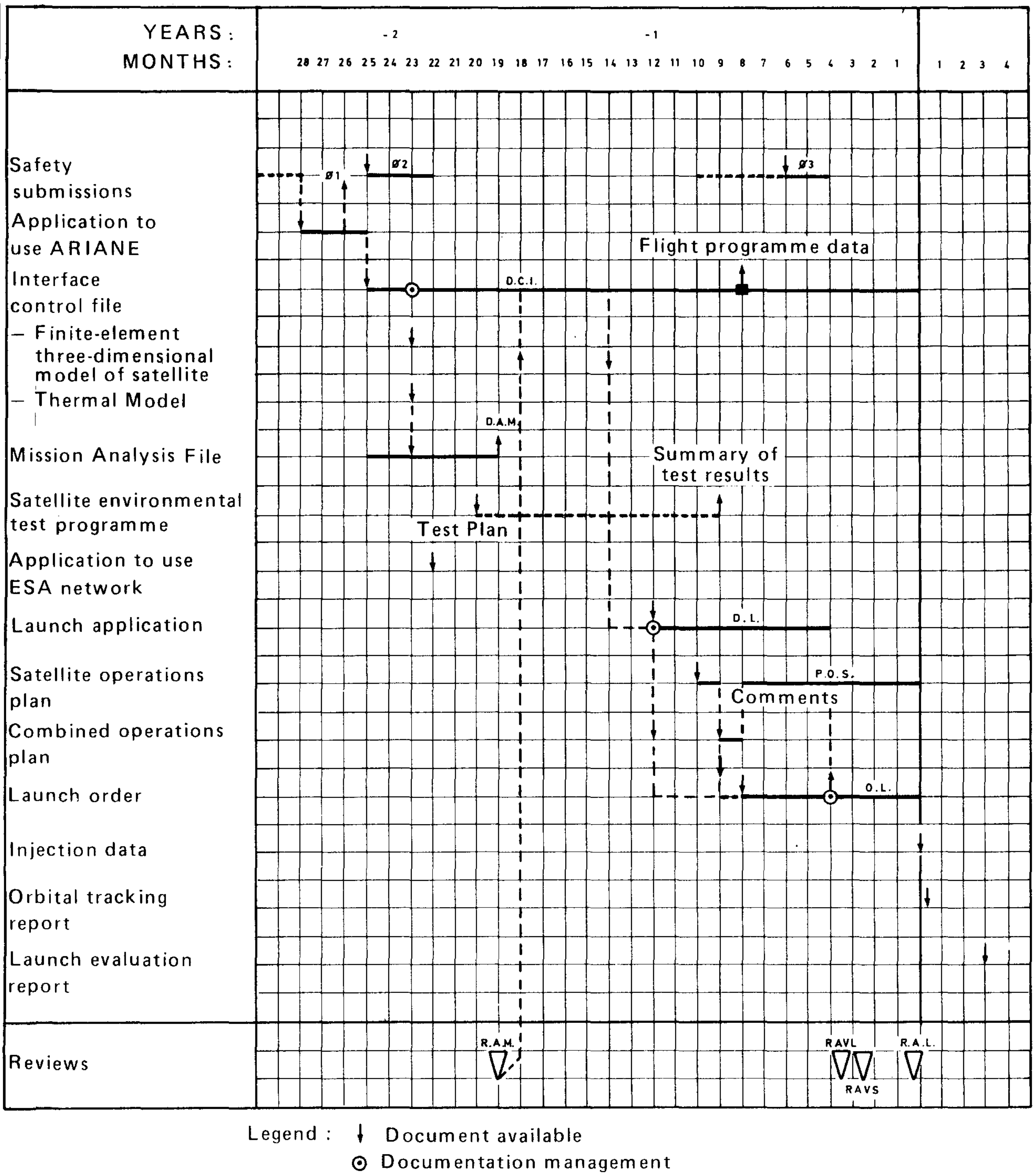


FIG. 6.2 TIME-SCHEDULE OF DOCUMENTATION AND REVIEWS

6.4. Format for application to use Ariane (DUA)

1. Introduction (brief description of user programme and description of satellite)
2. Number of launches and estimated dates
3. Mission requirements

3.1. Orbit requirements
3.1.1. Description of orbit
3.1.2. Injection accuracy

3.2. Launch vehicle/satellite separation requirements
3.2.1. Satellite system of axes
3.2.2. Satellite orientation
3.2.3. Spin-up
3.2.4. Sequence
3.2.5. Kinematic conditions

3.3. Launch windows
3.3.1. Solar aspect angle constraint
3.3.2. Elipse constraint
3.3.3. Constraint on ascending-node position
3.3.4. Constraint on inclination
3.3.5. Time-schedule of launch windows
3.3.6. Other constraints

3.4. Trajectory requirements
3.4.1. Thermal flux at fairing jettison
3.4.2. Roll manœuvres

4. Description of satellite

4.1. Mechanical interfaces
4.1.1. General view (configuration inside fairing) and dimensions
4.1.2. Ariane adaptor : details in the vicinity of the connector frame - rigidity at satellite rear frame
4.1.3. Non-standard adaptor : definition of adaptor and its interface with the launch vehicle. The user is required to define the adaptor supplied by him
4.1.4. Separation sensors
4.1.5. Satellite accessibility through fairing
4.1.6. Position of antennae and directivity

4.2. Masses, alignments, inertias (configuration inside fairing)
4.2.1. Estimated mass and tolerance values
4.2.2. Inertias on the three axes, with tolerances
4.2.3. Location of C G with tolerances
4.2.4. Description of sloshing fluid masses.
4.2.5. Dynamic balance with tolerances

4.3. Environmental requirements
4.3.1. Rigidities. Satellite fundamental frequencies when hard-mounted finite-element matrix-type three-dimensional satellite model
4.3.2. Acoustic level inside fairing

6.4. Format for application to use Ariane (DUA)

4.3.3. Acceptable temperatures during thrust phase. Power dissipated by satellite
4.3.4. Ground environment : cleanliness, humidity, ventilation temperatures
4.3.5. Elements sensitive to contamination

4.4. Electrical interfaces
4.4.1. Number and position of satellite umbilical plugs
4.4.2. Type of connector used
4.4.3. Electrical cable links between the satellite and the launch centre, at the latter and between the launch centre and the satellite checkout system
Links for use during satellite checks after erection, and checks in the course of the countdown
4.4.4. Radio links between the satellite, the launch centre and the satellite checkout system
For use outside and during countdown
4.4.5. Location and summary description of the satellite checkout system equipment in the launch centre and forward test post.
4.4.6. Continuity of Earth potential (potential reference)
- satellite reference point
- satellite checkout system (in Ariane launch-pad area), satellite mounted.

4.4.7. Links desired between the vehicle-checkout and satellite-checkout systems (cf. para. 3.5.2.3.)
4.4.8. Fairing radio-transparency requirements
4.4.9. Satellite transmit and receive systems
4.4.9.1. Description of systems
4.4.9.2. Satellite frequency plan
4.4.9.3. Utilization of systems on the ground and during launch
4.4.10. EMC characteristics - Standards used

5. Hazardous systems
(extract from Safety Regulations, para. 2.6.4.)

5.1. Electro-pyrotechnic devices
5.1.1. Category-A initiators (operation of which can be hazardous for personnel and equipment)
5.1.2. Category-B igniters (operation of which is not hazardous)
5.1.3. Location
5.1.4. Function
5.1.5. Type and manufacturer
5.1.6. Production serial number
5.1.7. Bridge resistance
5.1.8. No-fire current (associated probability and confidence interval)
5.1.9. Firing current (associated probability and confidence interval)
5.1.10. Selected firing current
5.1.11. Checkout current
5.1.12. Flegmatization current
5.1.13. Time required for installation on satellite
5.1.14. Location in satellite
5.1.15. Radio-sensivity characteristics
5.1.16. Electrostatic sensivity characteristics
5.1.17. Electrical initiation and control circuits

6.4. Format for application to use Ariane (DUA)

5.2. Other pyrotechnic devices :
5.2.1. Type and manufacturer
5.2.2. Characteristics
5.2.3. Arming and make-safe systems
5.2.4. Ignition systems
5.2.5. Will the satellite or one of its subsystems be shipped with the pyrotechnics mounted ?
5.2.6. Classification (see Safety Regulations, para. 4.1.1.3.)
5.2.7. Reference of qualification tests for classification
5.2.8. Is a waiver requested for the destruct system ?
5.2.9. Electrical initiation and control circuits

5.3. Liquid propellants
5.3.1. Do the payload and/or associated ground equipment contain hazardous fluids ? If so, indicate quantities and specifications.
5.3.2. Location and operating procedures

5.4. Pressurized gas
5.4.1. Do the payload and/or associated ground equipment contain pressurized gas ? If so, indicate quantity and specifications.
5.4.2. Are the CSG Safety Regulations complied with ?
5.4.3. Place and period of pressurization
5.4.4. Electrical and/or pneumatic initiation and control circuits

5.5. Radioactive products
5.5.1. Can the satellite and/or associated ground equipment emit ionizing radiation ? If so, indicate characteristics of the source.
5.5.2. Hazardous emission level (exceeding danger threshold)

5.6. Non-ionizing radiation
Can the satellite emit a hazardous level of non-ionizing radiation ?

5.7. Launch checkout and preparation facilities
5.7.1. Junction boxes, racks, (ground) sequencers used for the circuits referred to above, during preparation of the satellite before erection.
5.7.2. Junction boxes, racks, (ground) sequencers used for the circuits mentioned above, prior to launch.

5.8. Miscellaneous
5.8.1. Does the user have special requirements concerning the safety of persons and equipment ? If so, give details.
5.8.2. What precautions are taken for handling the satellite hazardous systems ? (max. temperature, shocks, etc.).
5.8.3. Are there any safety aspects not covered by the present questionnaire ? If so, give details.

6.4. Format for application to use Ariane (DUA)

6. Operational requirements
6.1. Requirements concerning CSG facilities needed for :
6.1.1. Transport and handling
6.1.2. Satellite checkout and integration hall (test facilities)
6.1.3. ABM preparation hall
6.1.4. ABM X-ray inspection hall
6.1.5. Satellite filling hall
6.1.6. Satellite/ABM integration hall
A diagram will be required, indicating satellite and pyrotechnic-element checkout and assembly logic
6.1.7. Launch site
6.1.8. Propellant and ABM storage (also satellite filling facilities)
6.1.9. Local communication and control facilities
6.1.10. Miscellaneous (chemical analysis, etc.)

6.2. Requirements for access to the satellite after integration with the launch vehicle
6.3. Constraints due to the satellite tracking network
Satellite configuration in transfer orbit, or after launch-vehicle/satellite separation
6.4. Provisional time-schedule for operation at the CSG

7. Satellite development plan
7.1. Development plan milestones
7.2. Requirements for launch-vehicle/satellite compatibility tests
7.2.1. Tests on the user's premises
7.2.2. Tests in France

8. Definitions

9. Annexes

6.5. Format for launch-system/satellite interface control file (DCI)

1. Introduction
2. Applicable documents
3. Document amendment procedure
3.1. Introduction
3.2. Procedure
3.3. Composition of Amendment Committee

4. Mission characteristics
4.1. Launch date
4.2. Launch window
4.3. Flight plan
4.3.1. Transfer orbit
4.3.2. Sequence of events
4.3.3. Kinematic conditions at separation

5. Launch-vehicle/payload interfaces
5.1. Payload configuration inside the fairing
5.2. Fairing/payload mechanical interface
5.3. Payload mass and inertia characteristics
5.3.1. Reference axes
5.3.2. Masses and tolerances
5.3.3. C of G and tolerances
5.3.4. Inertial characteristics and tolerances
5.4. Launcher-adaptor/payload mechanical interfaces
5.4.1. Mechanical interfaces
5.4.2. Assembly characteristics
5.4.3. Mechanical constraints
5.4.4. Assembly and access
5.4.5. Tools and templates
5.4.6. Compatibility tests
5.5. Ground environment inside fairing
5.5.1. Fluid interface
5.5.2. Temperature
5.5.3. Dissipated power
5.6. Flight environment (powered phase)
5.6.1. Static acceleration
5.6.2. Dynamic environment
5.6.3. Acoustic vibration
5.6.4. Thermal environment
5.6.5. Static pressure
5.6.6. Contamination
5.7. Launch-vehicle/payload electrical interface
5.7.1. Definition
5.7.2. Umbilical link
5.7.3. Signal characteristics
5.7.4. Allocation of umbilical and pad cables
5.8. Launch-vehicle and payload radio transmissions
5.8.1. Characteristics of radio systems
5.8.2. RF compatibility
5.8.3. Radio-system utilization slot
5.9. Electromagnetic compatibility

6.5. Format for launch-system/satellite interface

6.5. Format for launch-system/satellite interface

8. General facilities provided by the CSG
8.1. Transport and handling
8.2. Fluids and propellants
8.3. Safety Department support
8.4. Workshops and stores
8.5. Metrology and instrumentation
8.6. Telecommunications
8.7. Local communications and control facilities
8.8. Storage facilities
8.9. Accommodation and transport
8.10. Miscellaneous

9. Satellite test plan
9.1. Tests specific to the satellite
9.2. Launch-vehicle/satellite interface tests (and time-schedule)
9.2.1. Tests on the user's premises
9.2.2. Tests with launch vehicle (France)
9.2.3. Tests at CSG
- launch rehearsal
- compatibility with ground-station network

10. Documentation
10.1. General
10.2. Documentation specific to payload authority
10.3. Documentation specific to Ariane authority
10.4. Information after launch

11. List of reference documents

6.6. Format for satellite operations plan at CSG
("Plan des Opérations Satellite au CSG " - POS)

1. General
1.1. Introduction
2.2. Applicable documents

2. Schedules
2.1. Time-schedule
2.2. Table of weekly activities
2.3. Meetings - Organization - Contacts

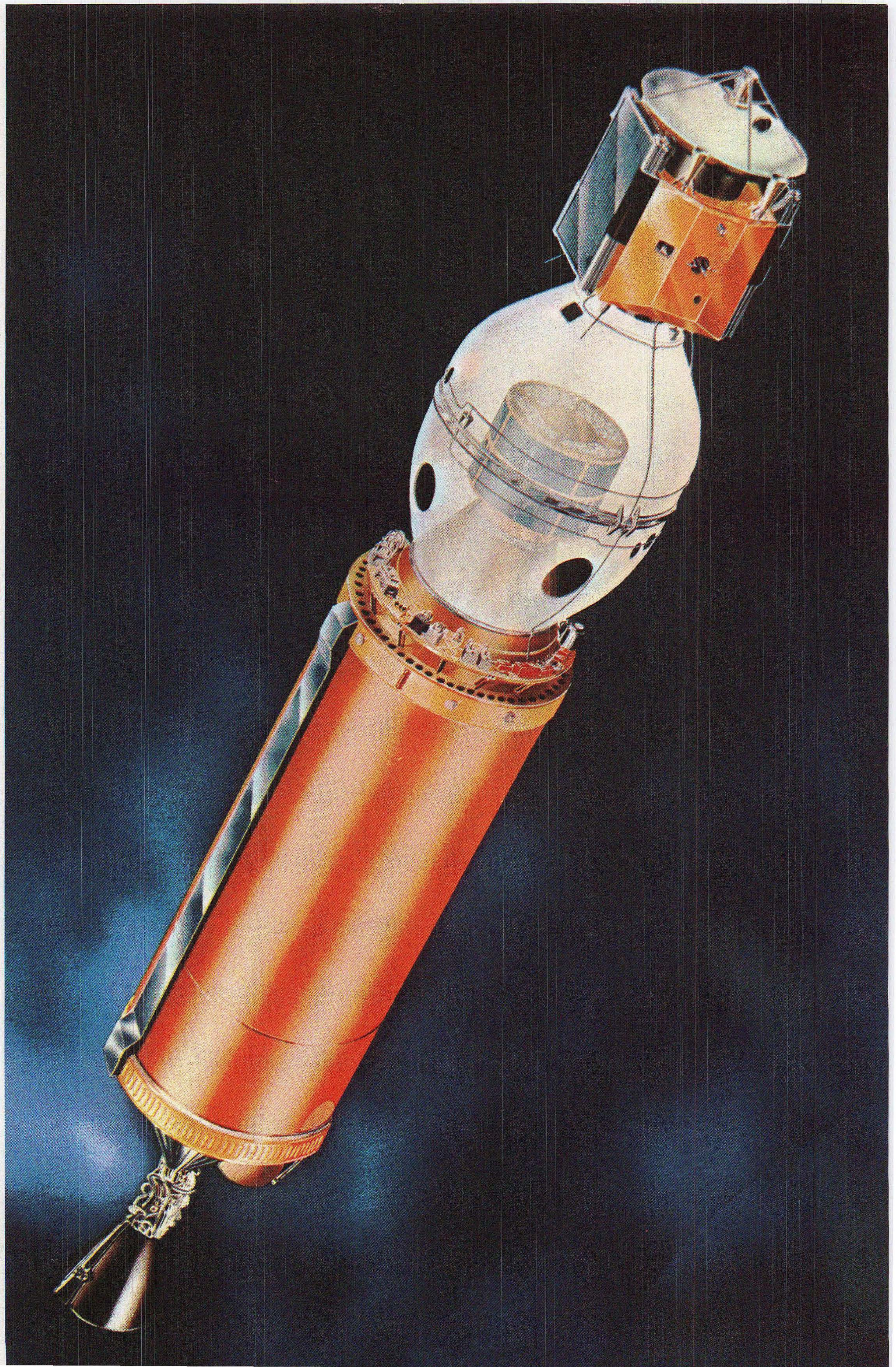
3. Personnel
3.1. Organization chart for satellite operations team
3.2. Definition of responsibilities and tasks
3.3. Organization chart for countdown

4. Operations
4.1. Handling
4.2. Tasks for launch operations

5. Equipment associated with the satellite
5.1. Brief description of equipment for launch operations
5.2. Description of hazardous equipment (with diagrams)
5.3. Description of special equipment (Launch Centre, servicing tower)

6. Installations
6.1. Surface areas
6.2. Buildings (technical and logistic aspects)
6.3. Intercommunication
6.4. Location of offices, assignment of personnel

7. Logistics
7.1. Accommodation
7.2. Transport facilities



Dual launch
artist concept showing: ARIANE 3rd stage
SYLDA
MARECS B
SIRIO 2 (inside SYLDA)

Dual launch system with Sylda — Chapter 7

7.1. Introduction

The Ariane launch vehicle can place simultaneously in orbit two independent satellites, the mass of which corresponds to that of satellites launched by the Delta vehicle or Space Shuttle with PAM-D (Payload Assist Module Class Delta). The device used for this purpose is a special adaptor designated Sylda ("Système de Lancement Double Ariane" - Ariane Dual Launch System).

The reference version of Sylda, which will be available as from early 1981, is compatible with the fairing and performance of the launch vehicle, as defined in the preceding chapters of this Manual.

It is intended to derive a " growth " version of Sylda from the reference version, at a later date. This version of Sylda will be designed for a longer fairing and a launch vehicle with improved performance, development of which is scheduled for 1980-83.

This chapter is intended to provide Sylda users with all the particulars specific to a dual launch that differ from those relating to single launches. All data relate solely to the reference version of the Sylda.

7.2. Description of Sylda

The Sylda consists of a load-bearing structure, comprising a conical adaptor carrying the lower satellite, and a shell, enclosing the lower satellite and carrying the upper satellite (see fig. 7.1). The shell comprises two separable parts, held together in flight by a clampband. Access to the lower satellite is via apertures provided in the shell for this purpose.

Radio transparency is ensured by a system of passive repeaters.

The attachment frames and separation systems for the upper and lower satellites are in principle identical, and the interfaces at these points correspond to those of the PAM-D. The two satellites have no mechanical or electrical interfaces with each other.

Injection and release of the two satellites is as follows. After 3rd-stage engine cut-off, the 3rd stage is oriented in the direction specified for the mission concerned, and (generally) spun up (max. 10 rpm). The 3rd-stage attitude-control system is then inhibited. Two explosive bolts are fired, releasing the clampband and separating the upper satellite. Separation velocity is imparted by four springs, bearing on the satellite. After

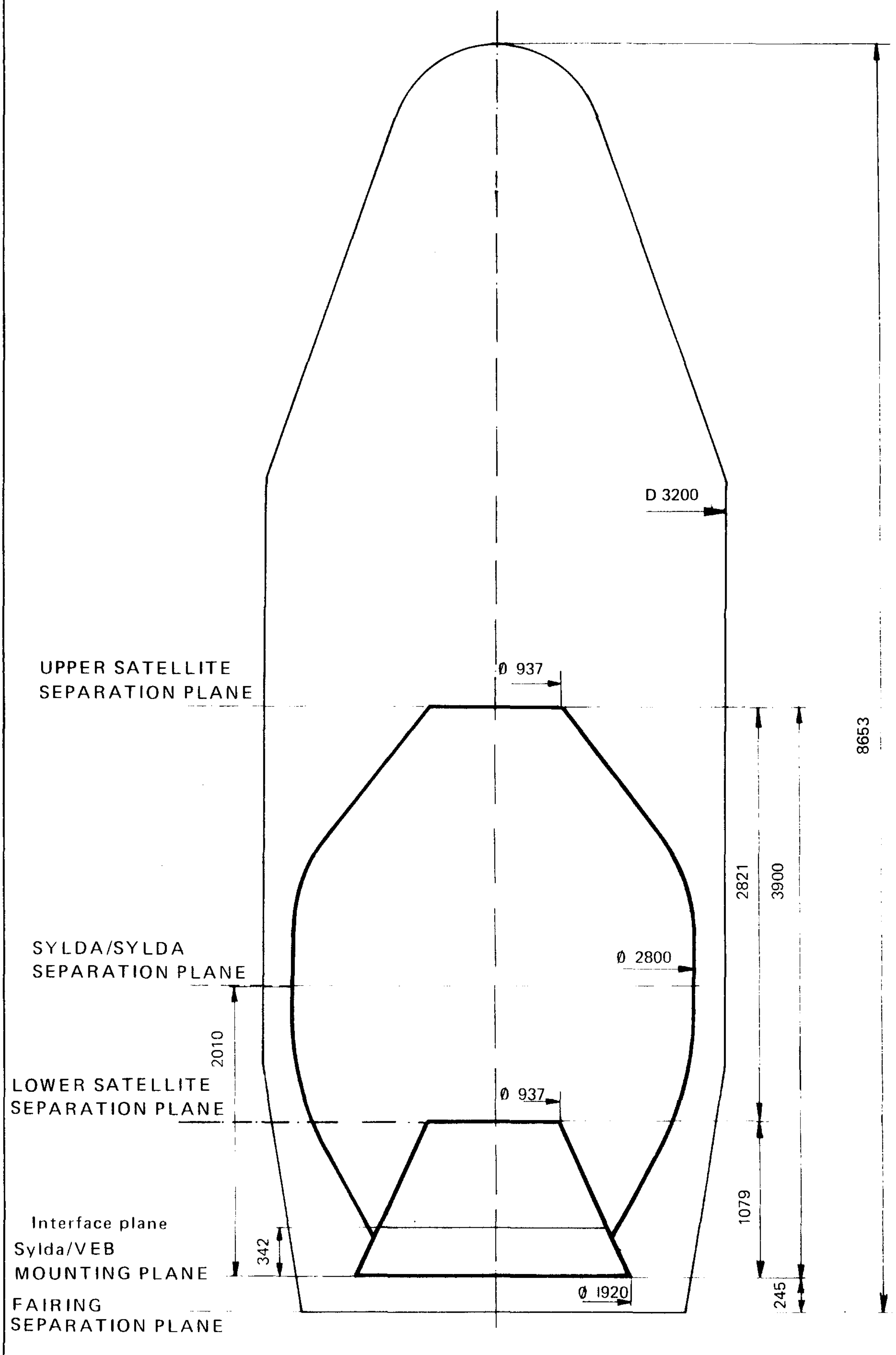


FIG. 7.1 GENERAL CONFIGURATION OF SYLDA BENEATH THE FAIRING

separation of the upper part of the support structure, the lower satellite is separated in the same way as the upper. Command signals for the three separation operations originate from the launch-vehicle VEB. Microswitches forming part of the Sylda equipment send telemetry signals confirming separation. The vibratory environment to which the two satellites are exposed in flight is measured and transmitted to the ground.

The relative velocities required for separation are determined individually for each mission, according to the characteristics of the satellites concerned, in order to avoid any risk of medium or long-term collision.

7.3. Sylda performance

7.3.1. Limitations applicable to the two satellites together

7.3.1.1. Mass

The general performance of Ariane when used for launching a single satellite mounted on the type 1194 adaptor is given in chapter 2.

Dual-launch performance can be deduced from single-launch performance by subtracting 140 kg from the latter. This reduction corresponds to the difference in mass between Sylda and the type 1194 adaptor, not used for a dual launch. Performance for a dual launch is defined as the sum of the masses of both satellites, including their ABMs.

7.3.1.2. Orbit

The two satellites are injected into almost the same transfer orbit, the slight difference being due to post-separation velocities differing by approximately 0.35 m/s.

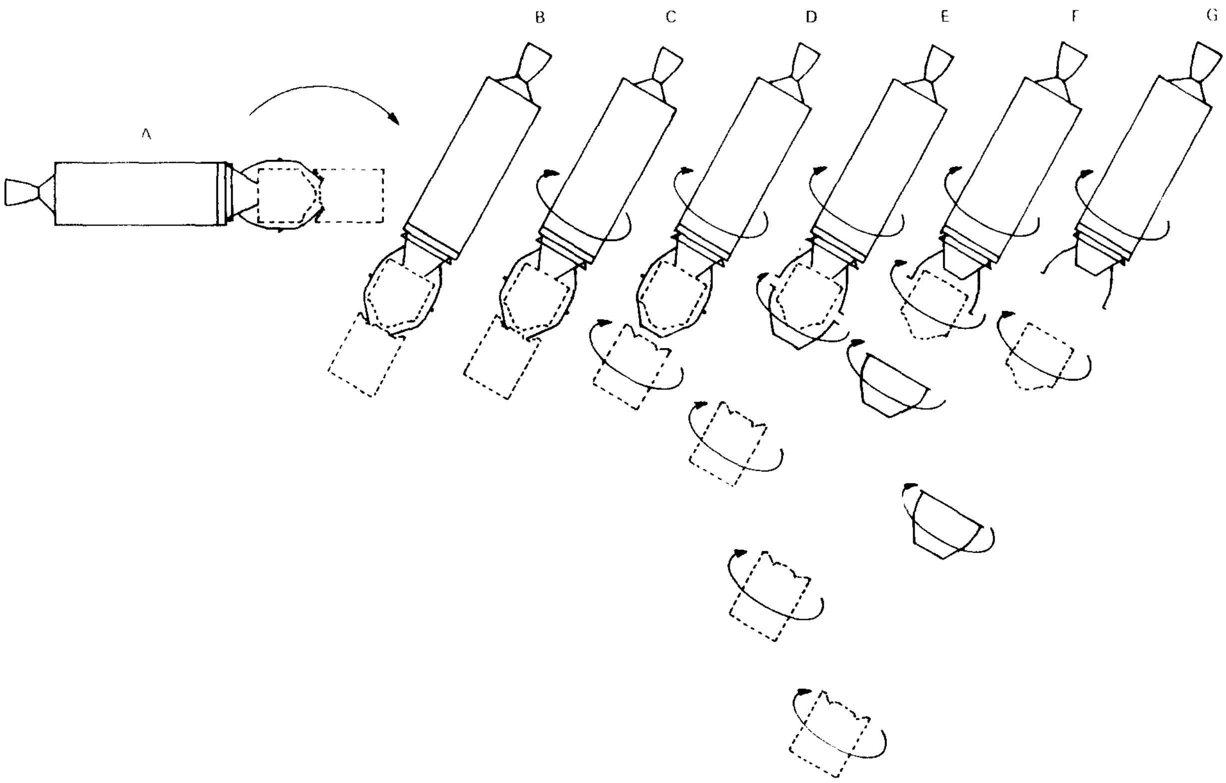
In general, the satellites are required to be placed in different final orbits, and different strategies are used for reaching final orbit from transfer orbit. Thus, the transfer orbit for a dual launch cannot be optimum for both satellites, and a **compromise transfer orbit must be found** at the feasibility study stage.

7.3.1.3. Reorientation and spin-up

The sequence of events after 3rd-stage thrust cut-off has been commanded by the on-board guidance computer (event " H3 ") is illustrated in detail in figure 7.2. Orientation and spin-up of the 3rd-stage and its payload are controlled by the 3rd-stage attitude and roll control system (" Système de Contrôle d'Attitude et de Roulis - SCAR ").

As the SCAR does not operate after spin-up has been completed, it is not possible to change either spin rate or orientation before the second satellite is separated.

A compromise must therefore be reached to select a common orientation, spin rate and spin direction for both satellites. While orientation



A - H3 THIRD STAGE CUT OFF COMMANDED
B - REORIENTATION OF COMPOSITE BY ATTITUDE CONTROL SYSTEM OF THIRD STAGE (SCAR)
C - END OF SPIN UP OF COMPOSITE BY ACTION OF SCAR NOT LESS THAN 150 SECONDS AFTER A SCAR OPERATION THEN INHIBITED
D - SEPARATION OF UPPER SATELLITE IMMEDIATELY AFTER C
E - SEPARATION OF UPPER PART OF SYLDA
F/G SEPARATION OF LOWER SATELLITE NOT LATER THAN 170 SEC AFTER A

FIG. 7.2 SEPARATION SEQUENCE

and spin direction can be selected according to satellite requirements, spin rate is largely determined by kinematic considerations (see para. 7.6).

7.3.2. Limitations applicable to each satellite

7.3.2.1. Envelope

The maximum permissible static envelope for the upper and lower satellites (in accordance with para. 7.8.1) is given in figures 7.3 and 7.4 respectively.

7.3.2.2. Mass and inertia characteristics

The Sylda design provides for the injection of two satellites, each of whose characteristics may vary widely, into a common (transfer) orbit.

The following table is a guideline, and indicates the ranges within which any combination of properties is acceptable.

Mass	600 to 1020 kg (1)
Moment of inertia about main axis	150 to 300 m^2 kg
Moment of inertia about transverse axis	110 to 300 m^2 kg
Height C G above Sylda/satellite separation plane	700 to 850 mm
Static unbalance	$\leqslant$ 1.3 mm
Dynamic unbalance	$\leqslant$ 0.25°
(1) Subject to overall mass limitation in para. 7.3.1.1	

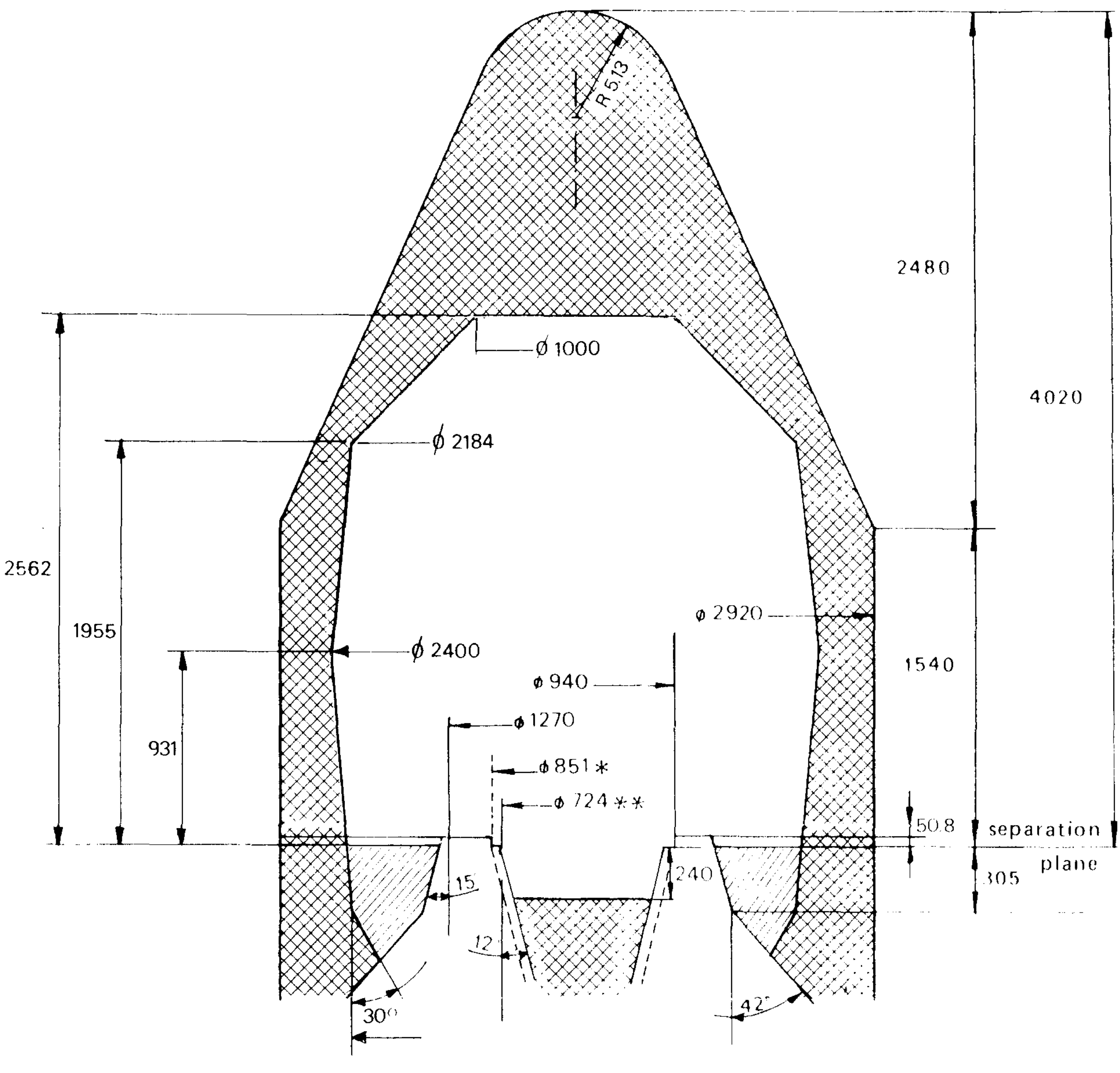
For characteristics outside the quoted ranges, dual-launch feasibility, which depends on the **combination of characteristics** of each satellite, and to a lesser extent on the combination of characteristics of both satellites together, must be assessed on a case-by-case basis.

The possible range for C G position is given in figure 7.5 as a function of mass.

See also para. 7.6 for possible range of static and dynamic unbalance characteristics.

7.3.2.3. Additional cylindrical adaptor

For satellites requiring an extension of the usable volume below the separation plane and within the cones subtended by the diameters 724 or 851, e.g. in order to accommodate apogee-motor nozzles, a cylindrical adaptor can be provided. It may be of any desired height, between 100 and 377 mm. Since the external profile of the volume usable by the satellite remains unchanged, the separation plane is thus moved up into the satellite. See figures 7.6 and 7.7. The adaptors remain attached to the Sylda structure, and separation takes place between the adaptor and the satellite. At the separation plane (top of the adaptor), the satellites



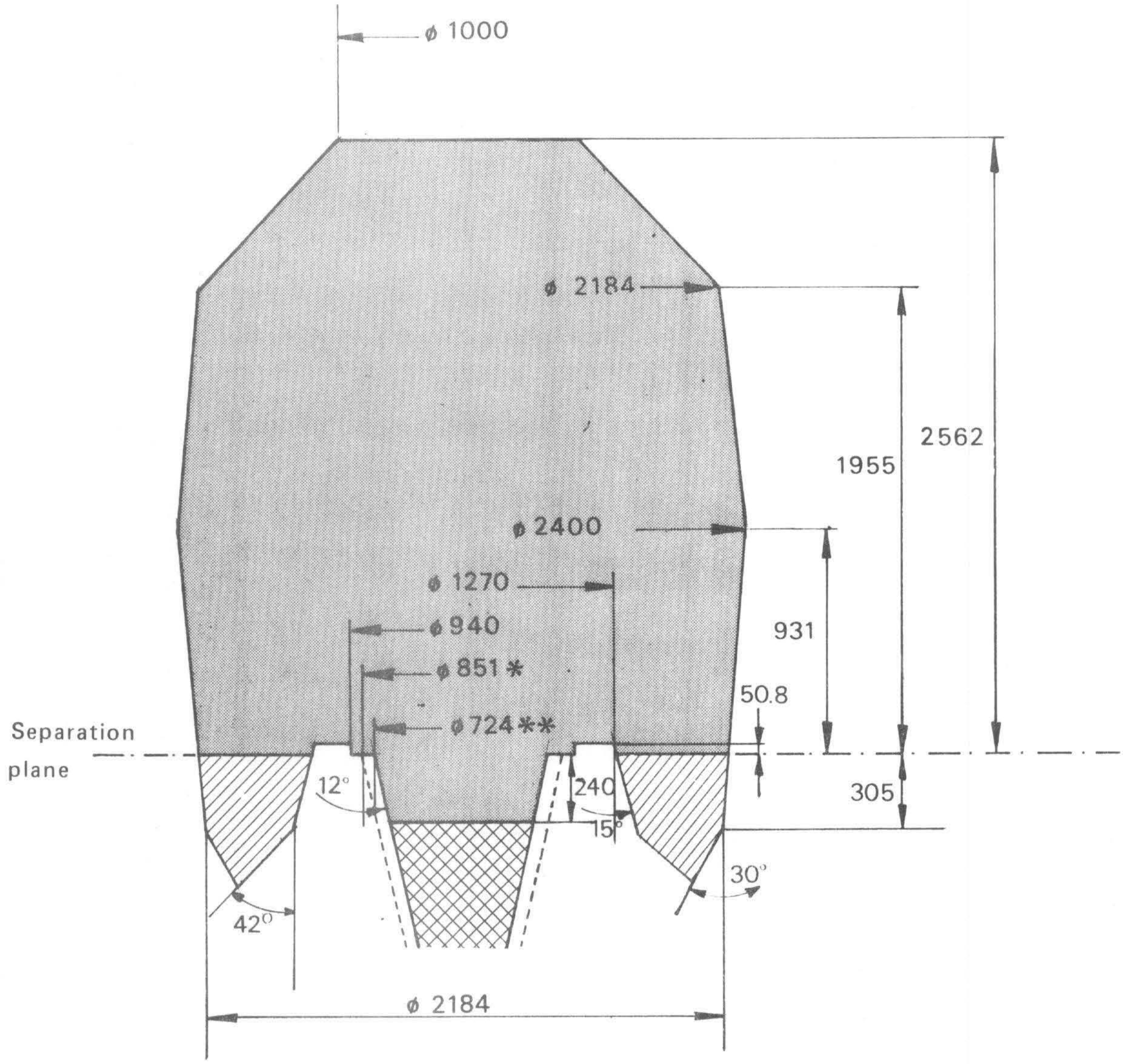
* External springs

** Internal springs

Volume subject to negotiation

Volume prohibiting switching from upper to lower position, and therefore subject to negotiation

FIG. 7.3 STATIC ENVELOPE, UPPER SATELLITE



* External springs

** Internal springs

Volume subject to negotiation

Volume prohibiting switching from upper to lower position, and therefore subject to negotiation

FIG. 7.4 STATIC ENVELOPE, LOWER SATELLITE

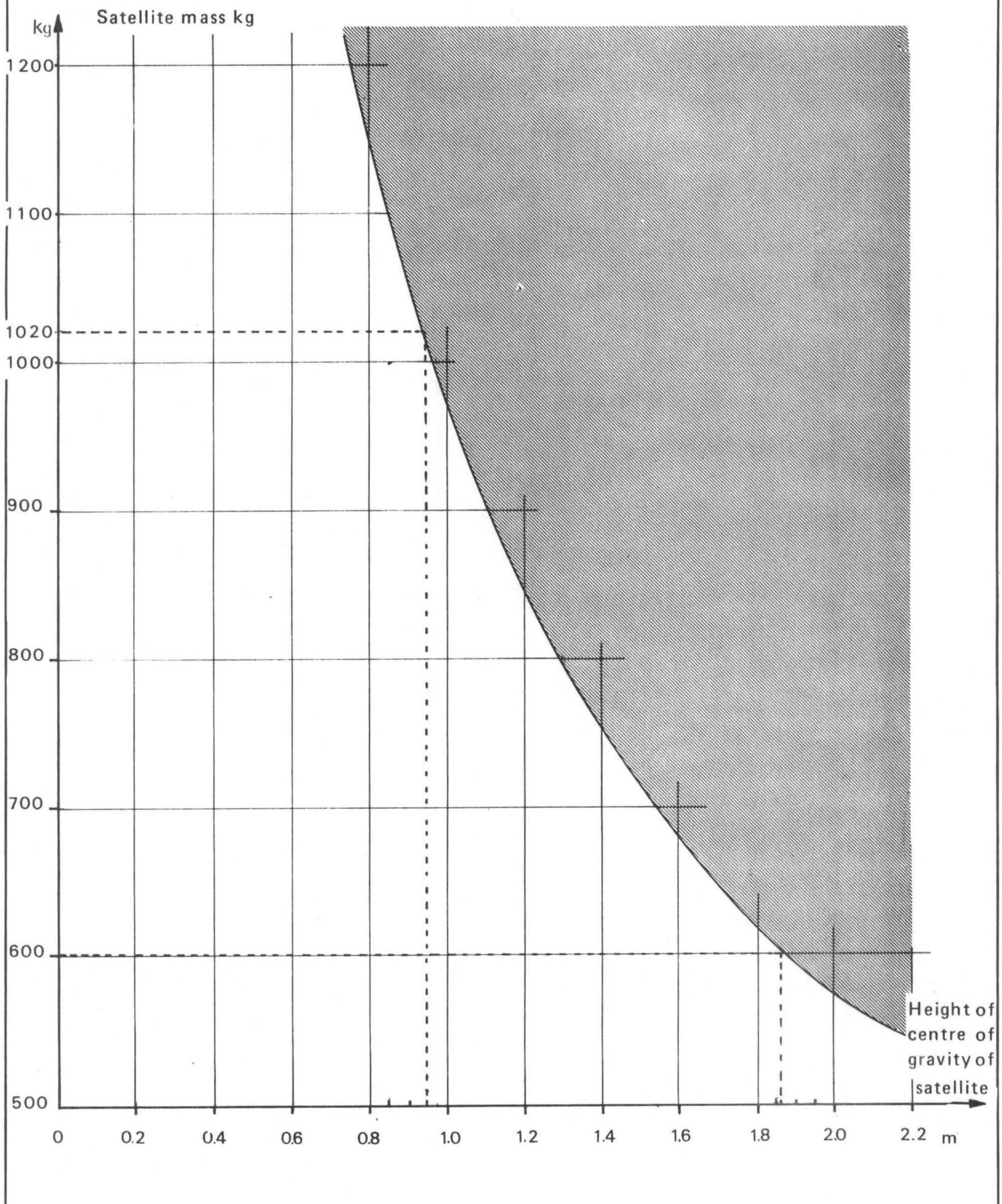
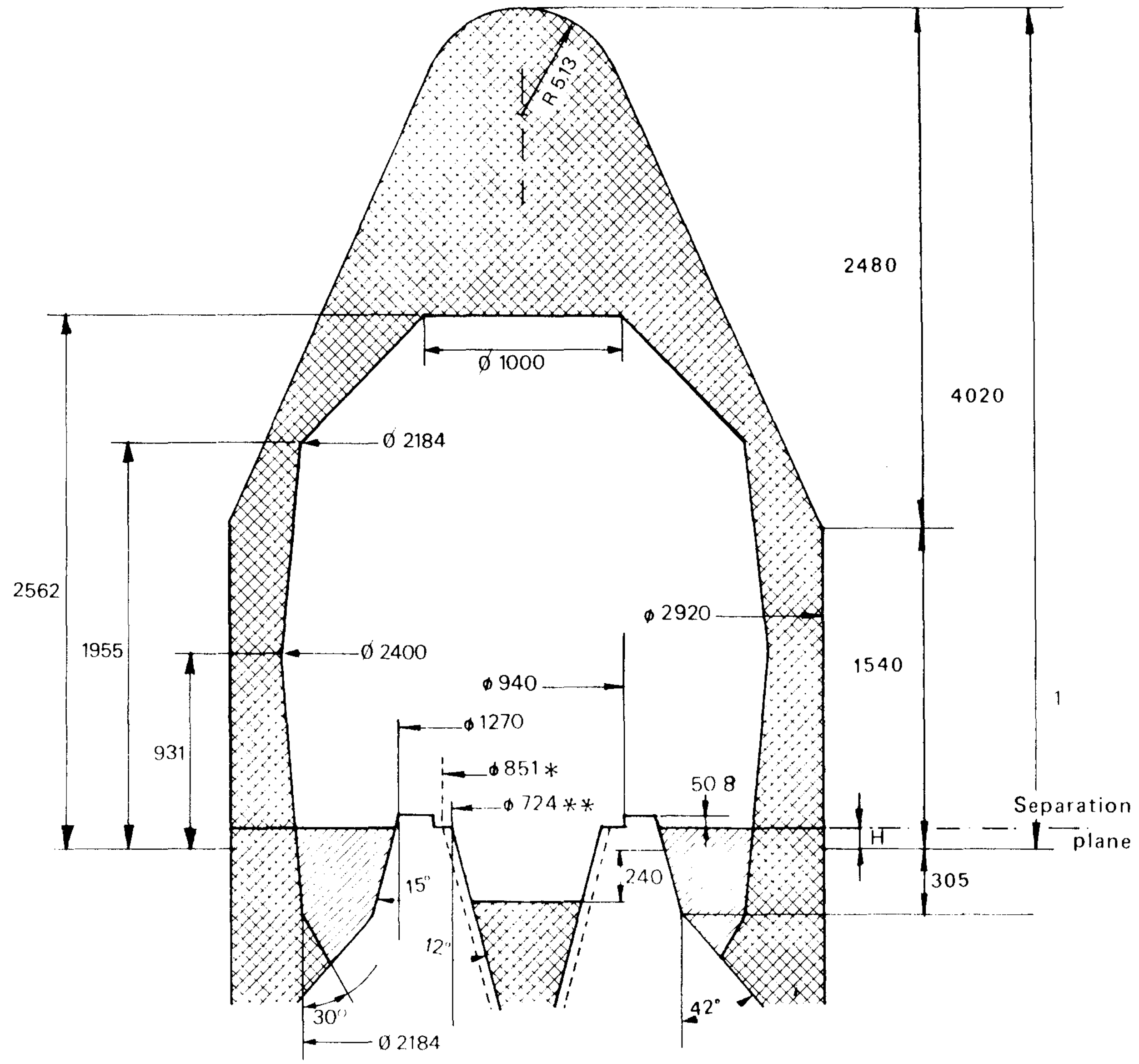
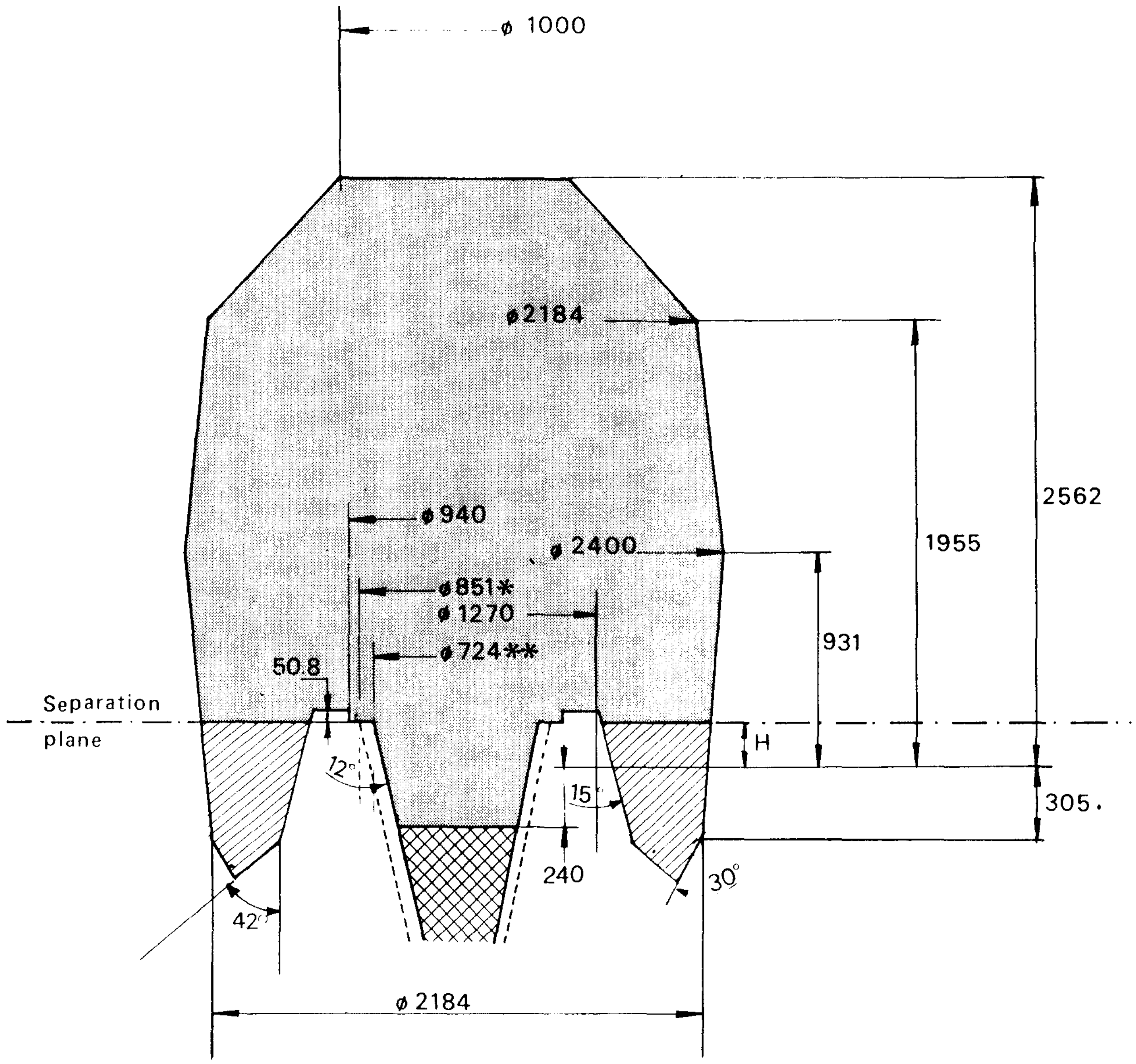


FIG. 7.5 CENTRE OF GRAVITY POSITION VERSUS PAYLOAD MASS



*	External springs
**	Internal springs
H	100 to 377 mm, height of adaptor
(single hatching)	Volume subject to negotiation
(cross hatching)	Volume prohibiting switching from upper to lower position, and therefore subject to negotiation.

FIG. 7.6 STATIC ENVELOPE, UPPER SATELLITE WITH SMALL ADAPTOR



*	External springs
**	Internal springs
H	100 to 377 mm, height of adaptor
(hatched)	Volume subject to negotiation
(cross-hatched)	Volume prohibiting switching from upper to lower position, and therefore subject to negotiation

FIG. 7.7 STATIC ENVELOPE, LOWER SATELLITE

encounter the same interfaces as when the Sylda is used without an adaptor ; see figures 7.8 and 7.9. The mass of the adaptor (TBD kg + TBD kg per mm of height) must be deducted from the mass available for the satellite. For the determination of the possible position of the satellite's centre of gravity, by means of figure 7.5., the reference remains the former separation plane (level 1079 or 3900 on fig. 7.1).

7.4. Mechanical interfaces

The mechanical interfaces between the upper and lower satellites and the Sylda are identical. Each satellite rests on a frame, the alignment diameter of which is 937.62 mm in the separation plane. The structures adjacent to these frames incorporate supports for the umbilical connectors and satellite separation and jettisoning systems.

Each satellite is anchored to its interface frame by a clampband. This comprises a metal strip applying a series of clamps to the payload and Sylda frames. Clampband tension does not exceed 20 000 N at any stage of the flight. The clampband assembly comprises two half-clampbands, connected by explosive bolts, the firing of which releases the clampband, which is then held captive by the Sylda assembly.

Attachment by the clampband introduces an angular flexibility of TBD rad/Nm at the Sylda/satellite interface.

The satellite is forced away from the launch vehicle by 4 springs, integral with the Sylda and bearing on supports fixed to the satellite rear frame. The force exerted on the satellite by each spring does not exceed 900 N. The positions of these springs can be selected by the user, inside or outside the satellite interface frame.

The Sylda interface frame is described in figure 7.8 (version with internal springs) and figure 7.9 (version with external springs).

Orientation of the satellite with respect to the launch vehicle is by a keyway corresponding to the launch-vehicle Z axis.

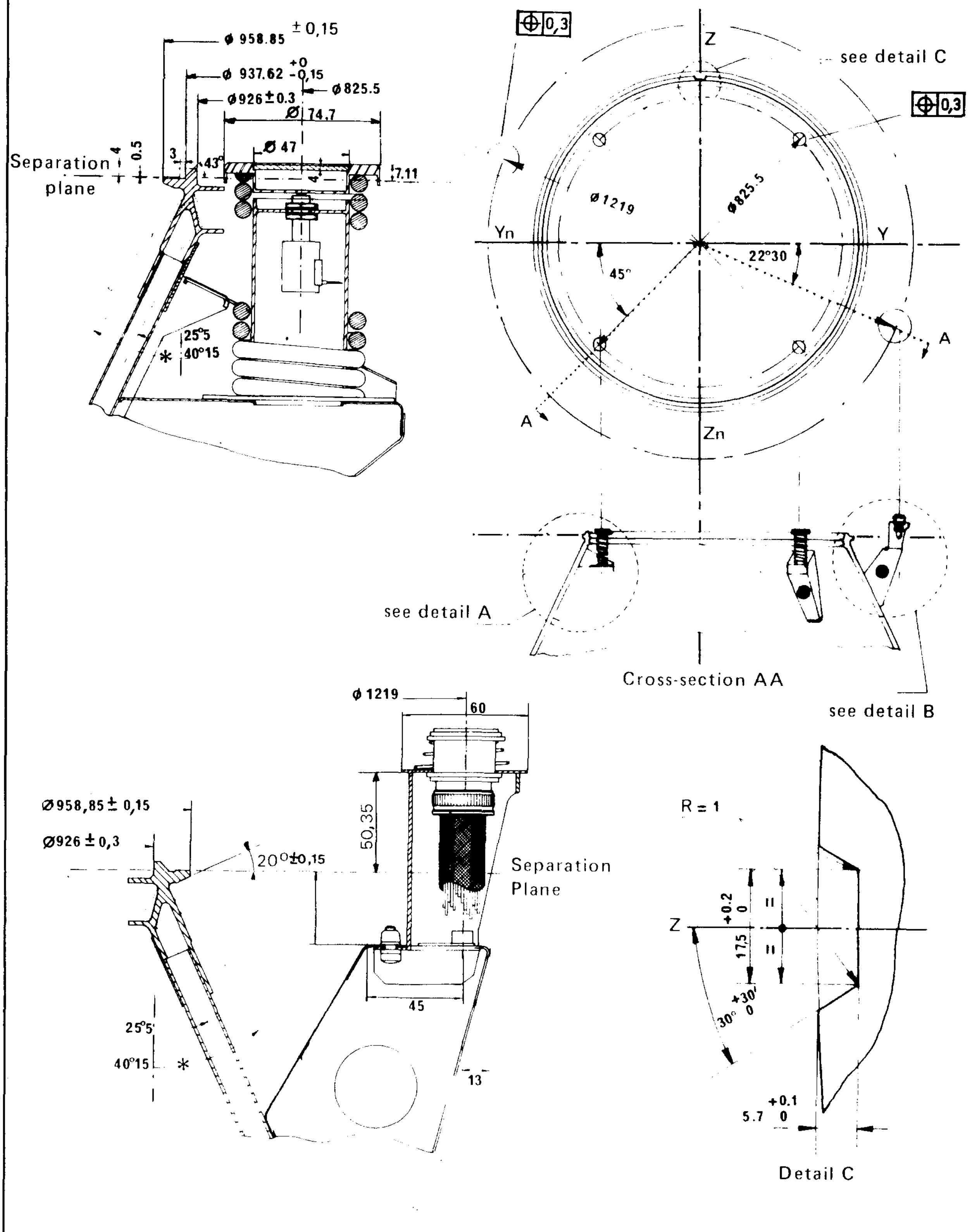
Figures 7.10 and 7.11 give the dimensions of the satellite rear frame and of the spring bearing surfaces, together with the positions of the umbilical plugs.

Two microswitches used to detect separation and located on the Sylda side inside the spring guides in the Y - Z and Y_N - Z_N quadrants (fig. 7.12). The Sylda assembly provides bearing faces for the satellite microswitches, aligned on the spring centre lines (fig. 7.13).

7.5. Satellite accessibility

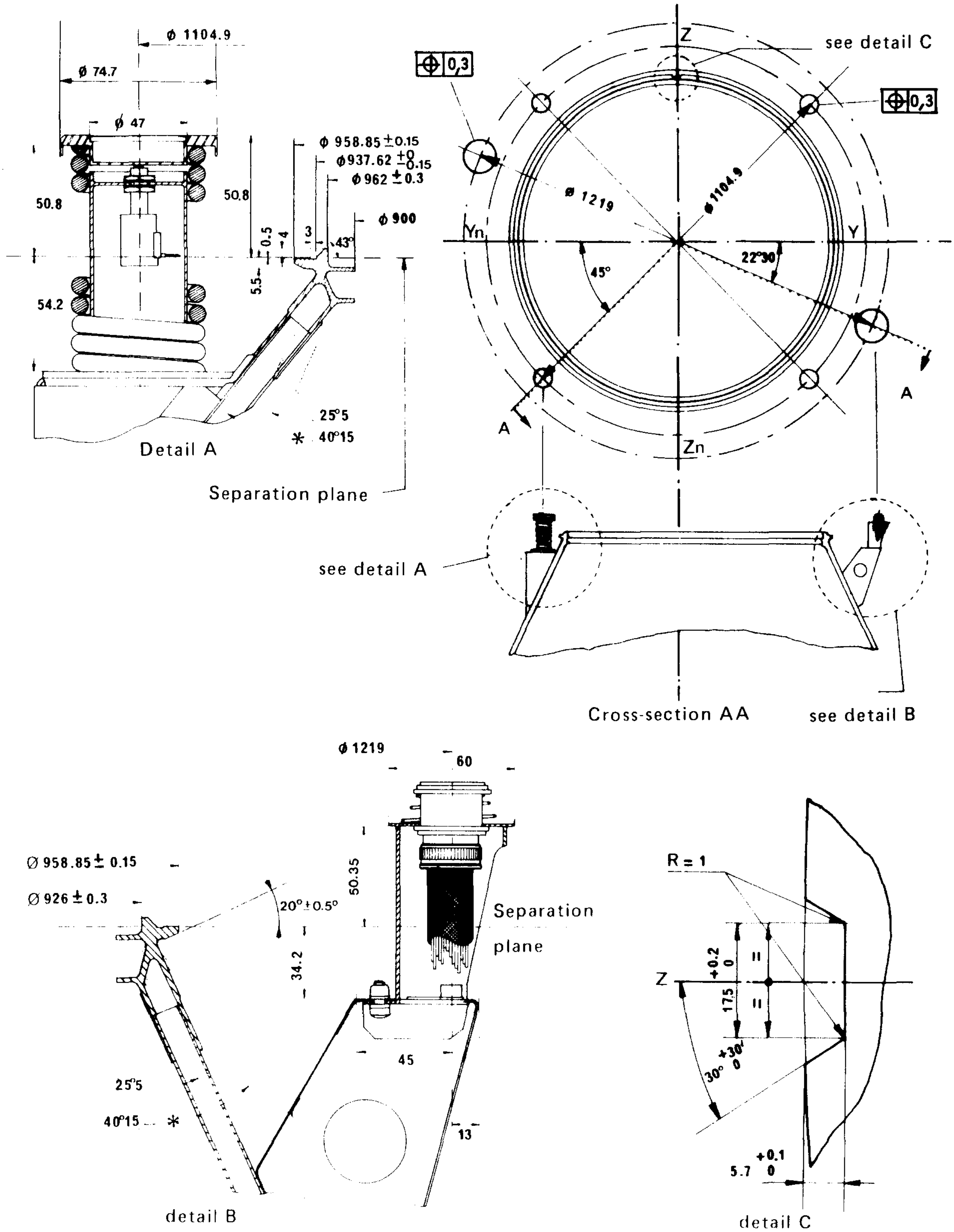
7.5.1. Access to upper satellite

Access to the upper satellite is via the fairing, by means of doors as described in para. 3.1.2.4.



* Upper section of SYLDA

FIG. 7.8. SYLDA FRAME, INTERNAL SPRINGS



* Upper section of SYLDA

FIG. 7.9 SYLDA FRAME, EXTERNAL SPRINGS

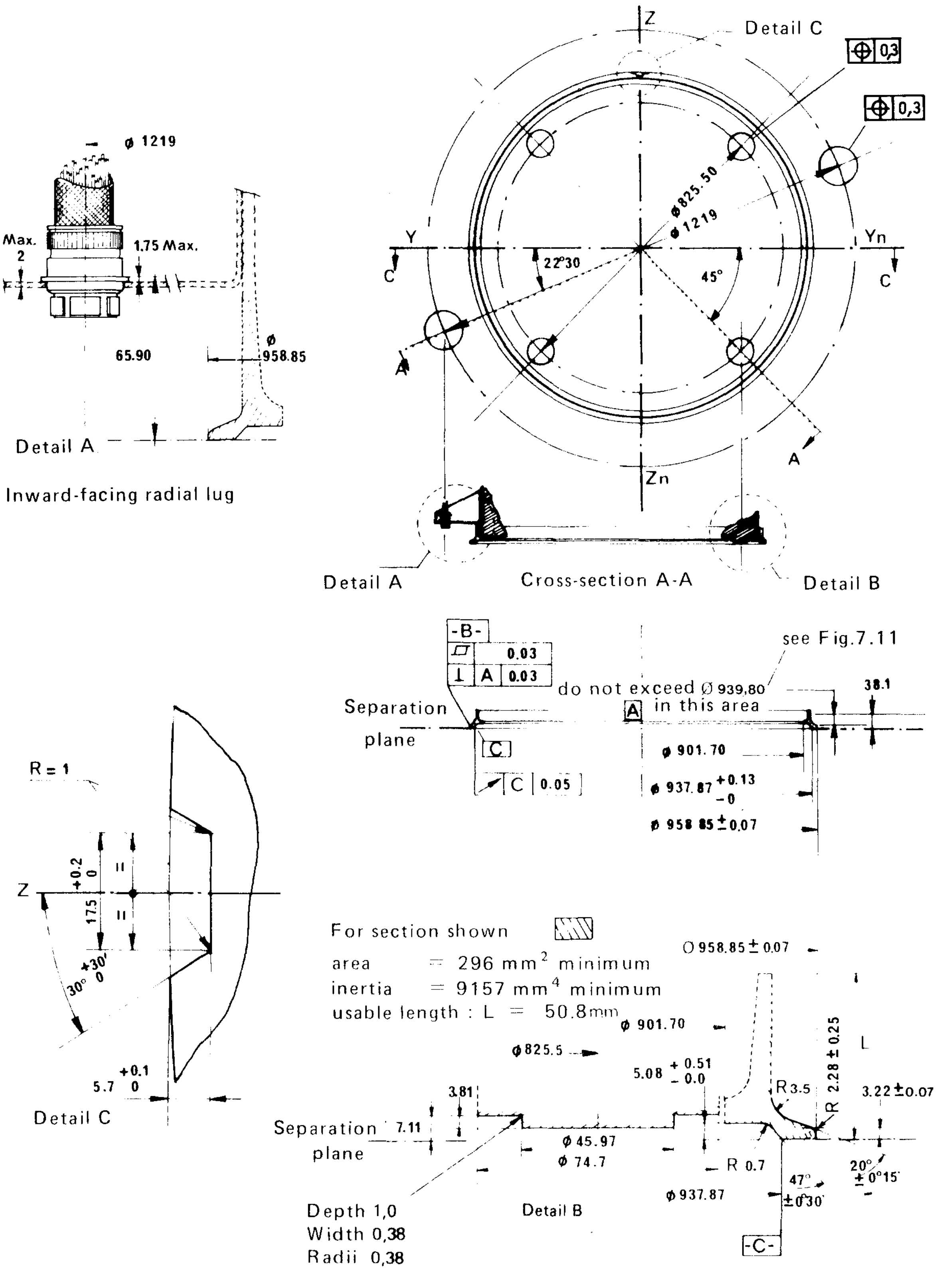
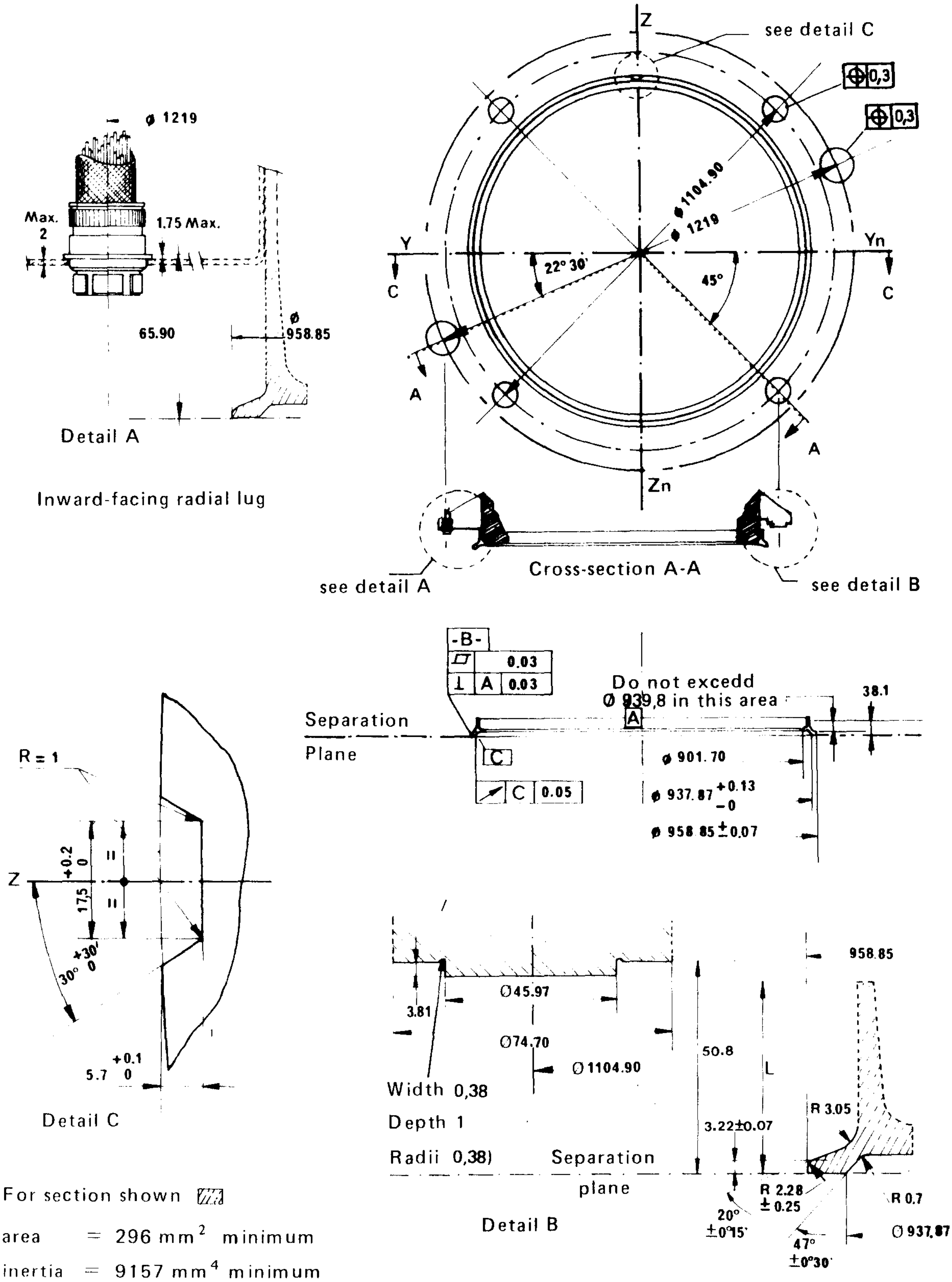


FIG. 7.10 SATELLITE REAR FRAME, INTERNAL SPRINGS



For section shown

area = 296 mm^2 minimum

inertia = 9157 mm^4 minimum

usable length L = 50.8 mm

FIG. 7.11 SATELLITE REAR FRAME, EXTERNAL SPRINGS

7.5.2. Access to lower satellite

Access to the lower satellite is via Sylda before fitting the fairing, and via Sylda and the fairing after the latter has been fitted.

Access possibilities via the fairing are described in para. 3.1.2.4.

Access via Sylda is provided by ports, determined for each mission according to the specific requirements of the payload within the following limits :

- max. number of ports : 4 ;
- dimension of each port : each aperture required must be inscribed within a 200 × 200 mm square ;
- the distance between the centres of the circumscribed circles for two adjacent ports must be greater than the sum of the diameters of these circles ;
- location areas for the ports are shown in figure 7.14. Their exact position are determined by agreement with the Ariane authority.

Any request falling outside the above limits will be subject to a special feasibility study.

7.5.3. Constraints on access via the fairing

(umbilical access)

To achieve optimum positioning of the door required for the Sylda and the access doors required for each of the two satellites, the total number of doors and their location will be defined, and may be the subject of a compromise.

7.6. Kinematic conditions at separation

7.6.1. Separation with spin-up

Kinematic conditions at separation are expressed in terms of the following two quantities :

δ_H : pointing error of the kinetic moment vector
ω_T : transverse angular velocity due to nutation

The relationship between these two quantities and the nutation-cone half-angle is shown in figure 7.15.

The following table gives values for δ_H and ω_T which will not be exceeded with a probability of 99.73 %. The corresponding nutation-cone half-angle values Θ for satellites with a transverse moment of inertia equal to the axial moment of inertia (inertial ratio 1) are also given.

	δ_H	ω_T	Θ $I_R/I_T = 1$
Upper satellite	5.33°	1.60°/s	1.53°
Lower satellite	5.55°	1.96°/s	1.87°

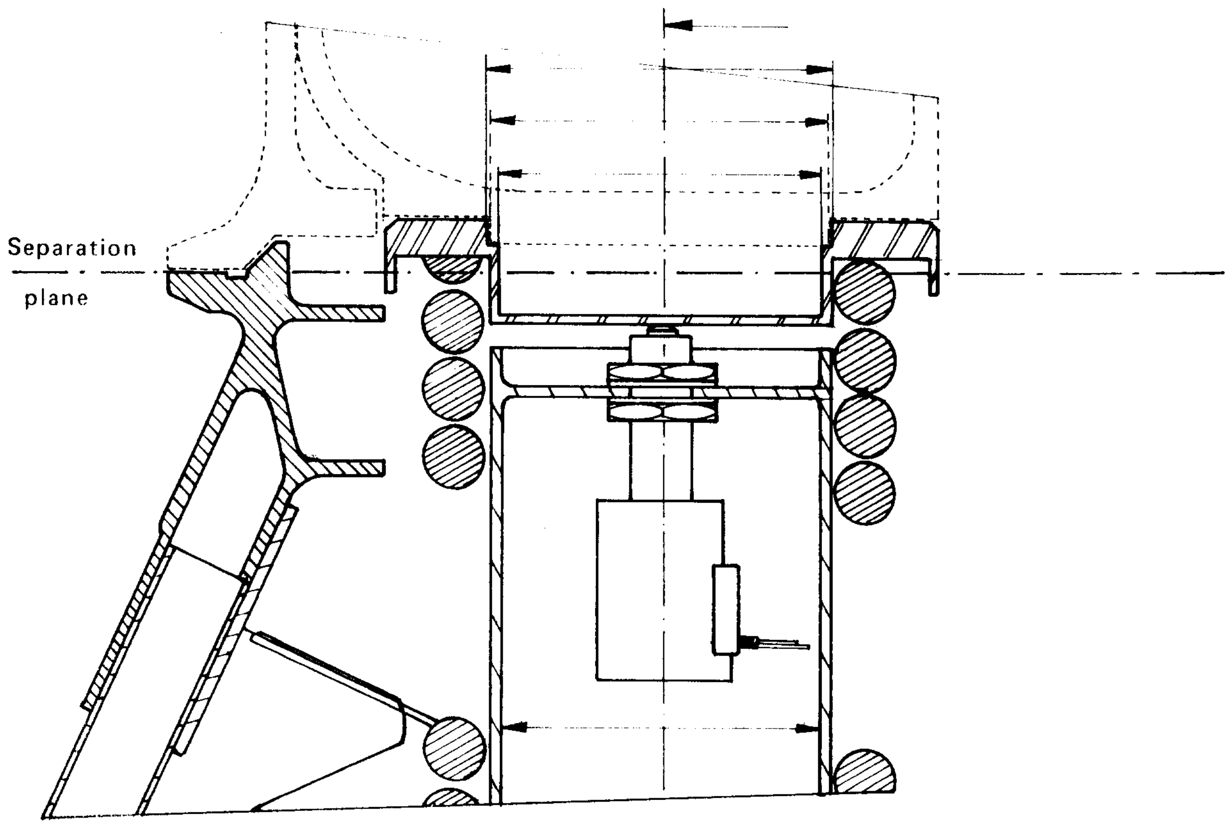


FIG. 7.12 SEPARATION MICROSWITCH INSTALLATION IN SYLDA SEPARATION SPRINGS
(SHOWN FOR SPRINGS IN INTERNAL POSITION)

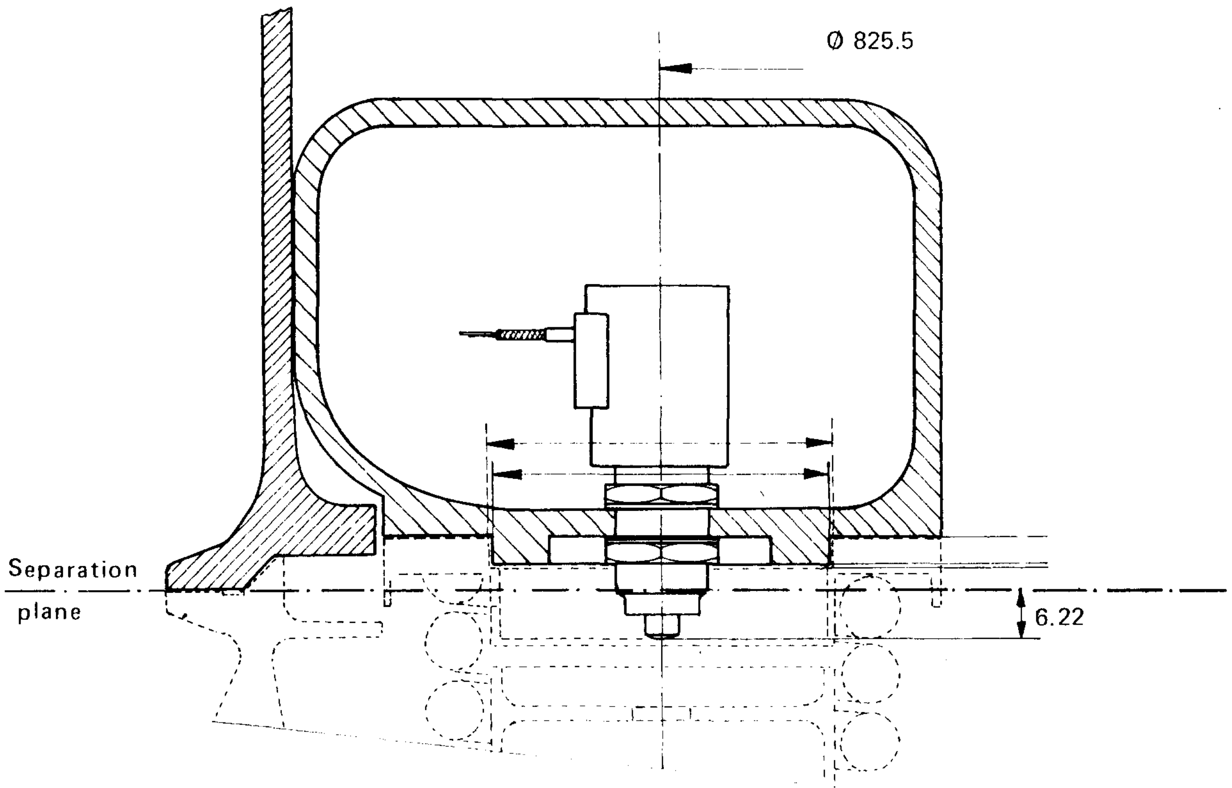


FIG. 7.13 TYPICAL SEPARATION MICROSWITCH INSTALLATION IN SATELLITE
(SHOWN FOR SPRINGS IN INTERNAL POSITION)

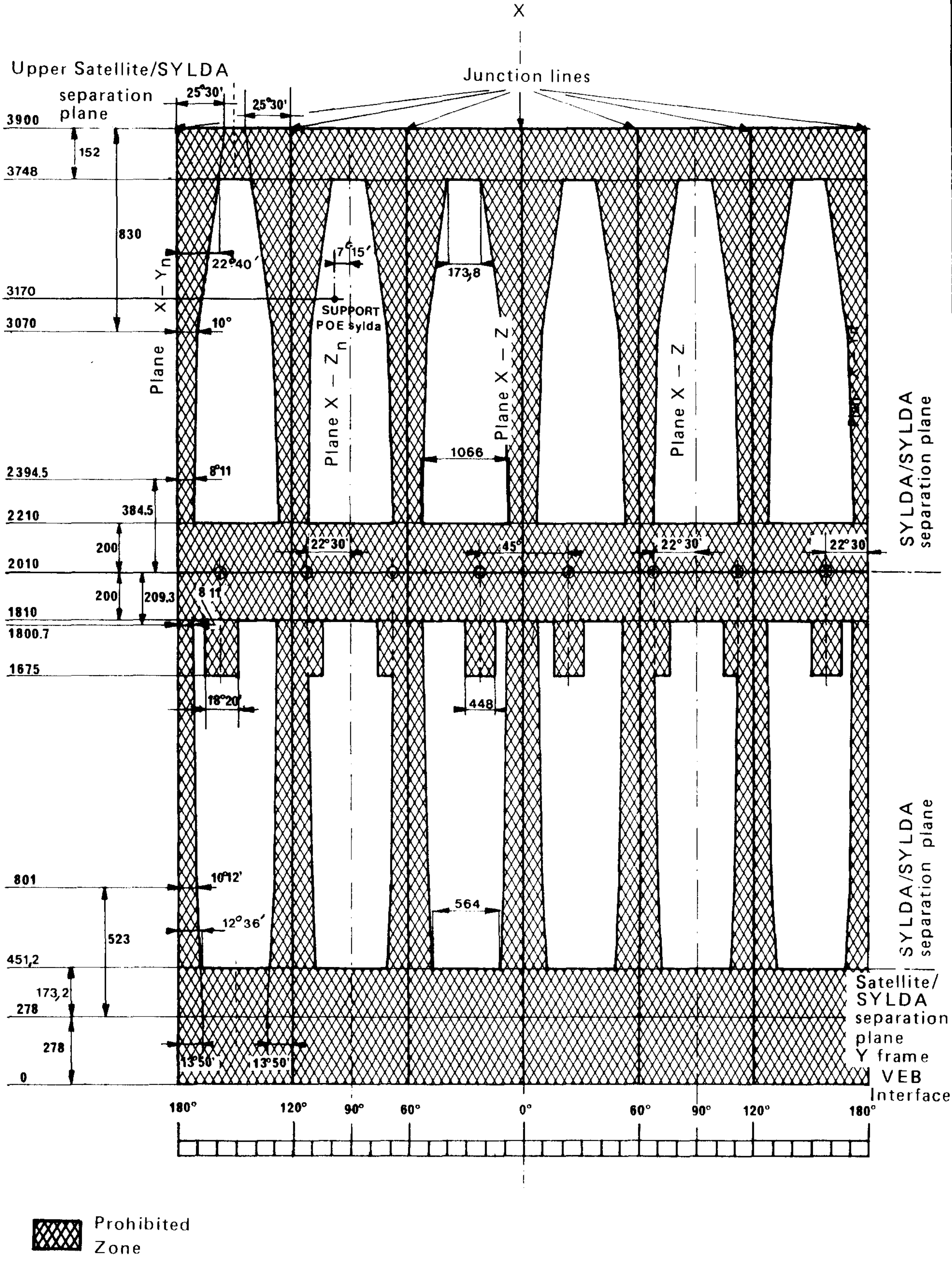


FIG. 7.14 POSSIBLE LOCATION ZONES FOR ACCESS WINDOWS THROUGH SYLDA
Cylindrical projection

The table is applicable when the maximum spin rate (10 rpm) that can be provided by the SCAR is used, and for satellites with characteristics falling within the ranges shown in the table in para. 7.3.2.

It should be noted that the worst possible case was assumed for establishing the results, with maximum satellite mass of 1020 kg associated with the highest C G position, and the lowest possible moment of inertia (150 m^2/kg) in both the transverse and axial directions. In general, therefore, for satellites with characteristics within the ranges given in the table in para. 7.3.2., pointing error and transverse angular velocity values will be less for the same probability of 99.73 % than those shown in the table.

For satellites with characteristics outside the range considered, or requiring a lower spin rate, calculations must be made case-by-case. It should also be noted that satellites with selected characteristics greatly exceeding the ranges considered will not necessarily be separated under substantially less favourable kinematic conditions.

For example, doubling of the effect of dynamic unbalance of the upper satellite increases the value of ω_T for this satellite by only 5 %, and has no significant effect on pointing error.

Lower spin rates may be possible, if higher depointing and nutation angles are acceptable for both satellites.

7.6.2. Separation without spin-up

Separation at zero spin can also be considered. Details TBD.

7.7. Environment

7.7.1. Mechanical environment

7.7.1.1. General

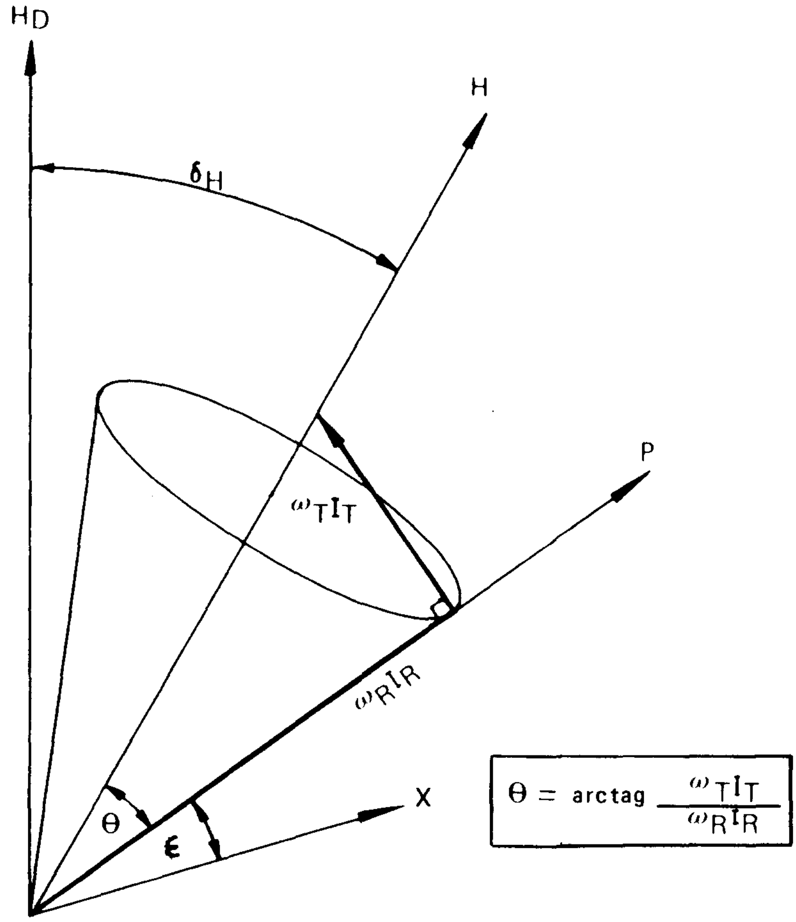
The payloads are subjected to static and dynamic loads in flight, induced by the launch vehicle.

The sources of excitation are described in para. 3.4.1.1.

In the following paragraphs, the base of each payload is defined as being the top of the associated adaptor.

7.7.1.2. Static acceleration

See chapter 3.4.1.2.



H_D	required direction of Moment of Inertia vector
H	actual direction of Moment of Inertia Vector
P	satellite principal axis of inertia
X	satellite geometric axis
δH	pointing error of Moment of Inertia Vector
θ	nutation cone half-angle
ξ	satellite dynamics umbalance angle
ω_T	transverse angular velocity
ω_R	spin angular velocity
I_T	transverse Moment of Inertia
I_R	axial Moment of Inertia

FIG. 7.15 KINEMATIC CONDITIONS AT SEPARATION

7.7.1.3. Dynamic environment

7.7.1.3.1. Longitudinal sinusoidal vibration

Vibration levels at the base of the satellites correspond to the excitation values given in para. 3.4.1.3.1.

The spectrum at the base of the lower satellite is 1.5 g from 10 to 100 Hz.

The spectrum at the base of the upper satellite allows for an increase in level in the 18-28-Hz band, resulting from Sylda resonance (2.3 g instead of 1.5 g). The spectra are shown in figure 7.16.

7.7.1.3.2. Lateral sinusoidal vibration

Vibration levels at the base of the satellites depend on the flight instants concerned, and are given in para. 3.4.1.3.2. Below 10 Hz, the first lateral mode of the Sylda is coupled with the first bending modes of the launch vehicle, leading to higher levels at the base of the upper satellite (2 g instead of 1.5 g).

Lateral sinusoidal vibration spectra are given in figure 7.17.

7.7.1.3.3. Random vibration

Defined in para. 3.4.1.3.3., and applicable to both payloads.

7.7.1.3.4. Acoustic vibration

Acoustic vibration inside the fairing is defined in para. 3.4.1.3.4. It should be noted that acoustic insulation can also be used for dual launches. Acoustic vibration level inside the Sylda (in both cases) : TBD.

7.7.1.3.5. Shock

The upper satellite is subjected to shocks principally at separation.

The lower satellite is subjected to shocks principally at fairing jettison, upper-satellite separation, jettison of the upper part of the Sylda and at its own separation.

The frequency content of these excitations is over TBD Hz.

7.7.2. Thermal and climatic environment

The environment in the payload facilities, inside the payload container, and in the servicing tower without the fairing is defined in para. 4.6.

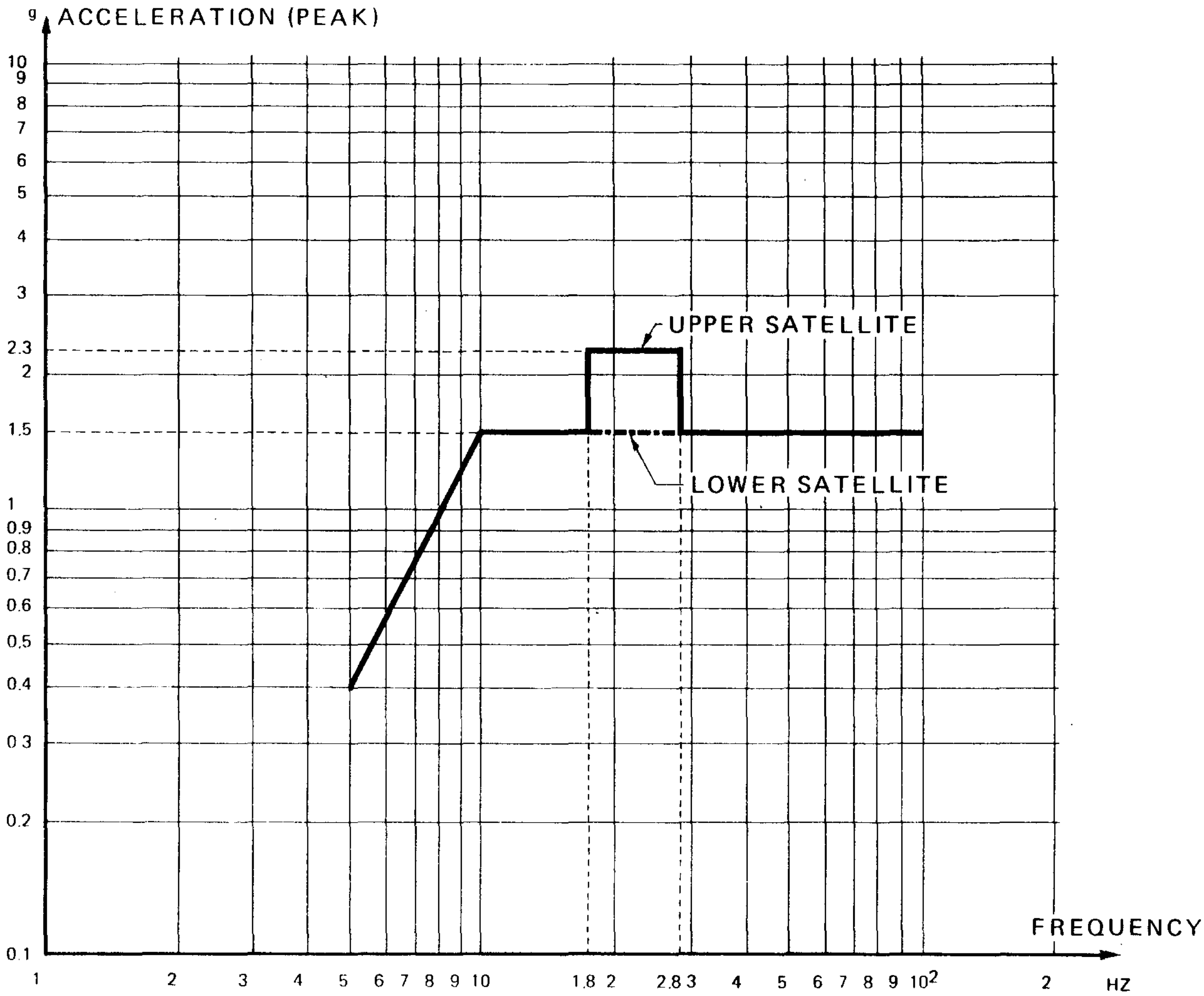


FIG. 7.16 LONGITUDINAL SINUSOIDAL VIBRATIONS (FLIGHT LEVELS)

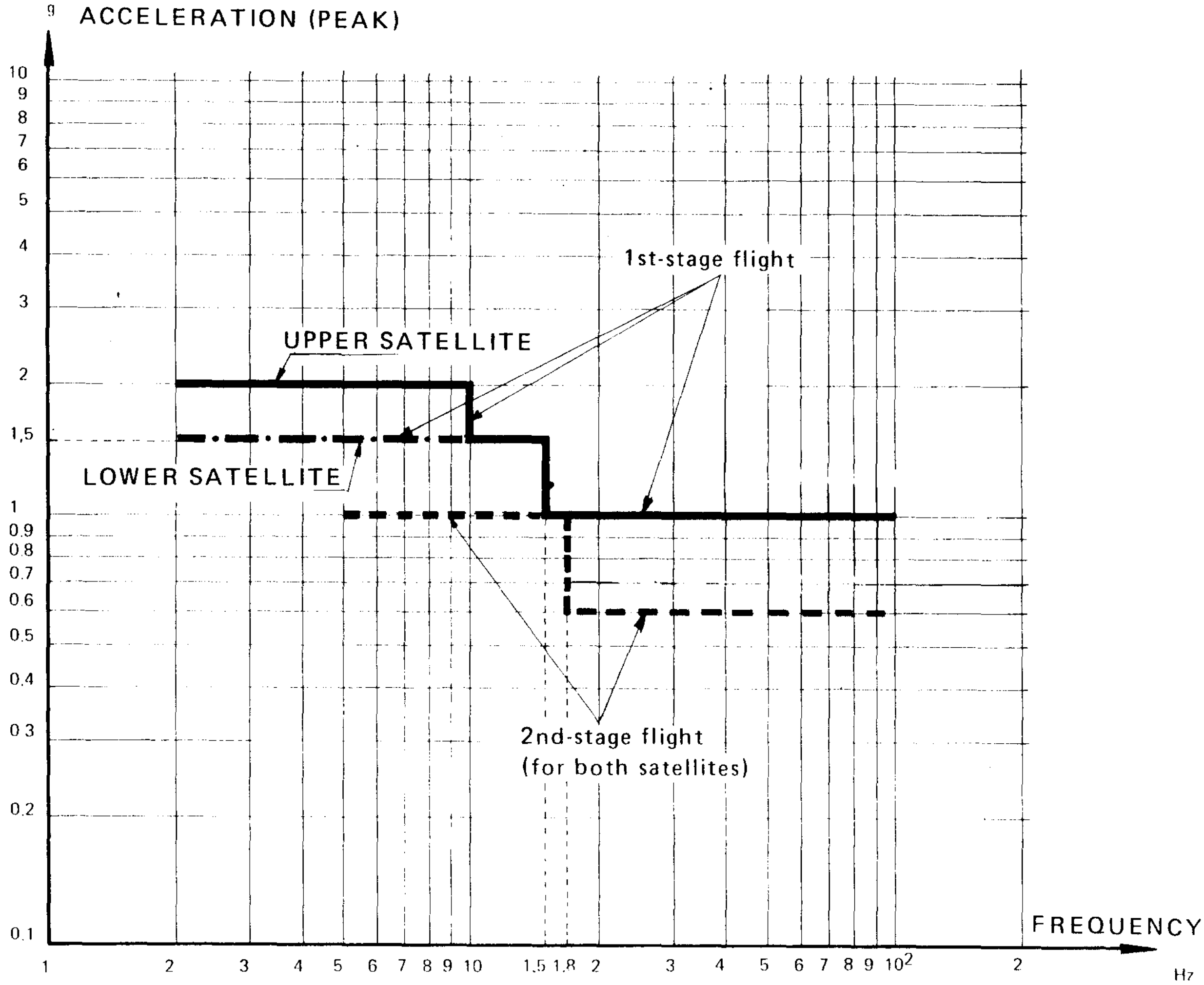


FIG. 7.17 LATERAL SINUSOIDAL VIBRATIONS (FLIGHT LEVELS)

Dual launch system with Sylda — Chapter 7

Ground environment inside the fairing is described in para. 3.4.2.2. During this phase, the temperature of the Sylda does not exceed 35° C, irrespective of waiting conditions, but subject to a maximum satellite dissipation of 100 W. The Sylda thermo-optical characteristics are as follows :

- outer face absorptivity α = TBD
- outer face emissivity ε = TBD
- inner face emissivity ε = 0.78

Ventilation of the lower satellite is by : TBD.

The evolution of Sylda temperature in flight is shown in figure 7.18, for different points of the structure and for day and night launch conditions.

7.8. Concept and dimensions

7.8.1. Frequency requirements

To avoid dynamic coupling between the launch vehicle and satellites, the following measures are called for :

- the fundamental bending-mode frequency of each satellite, assumed to be hard-mounted, should be at least 15 Hz and preferably higher.
- the frequency of the principal longitudinal modes should be 40 Hz or more, and within the mass/frequency domain defined in figure 7.19.

It should also be noted that transient thrust-decay excitations in the 35-50-Hz band may excite secondary structures and flexible elements (antennae, solar panels, etc.), the definition of which should ensure that their fundamental frequencies fall outside this band.

7.8.2. Primary-structure dimensioning loads

The various static and dynamic loads are superimposed during the flight. The design and dimensioning of the primary structures of each payload must therefore allow for the most stringent load combinations likely to be encountered at any time during flight. From the point of view of quasi-static loads, conditions are most stringent at the following times :

- maximum dynamic pressure
- first and second-stage burn-out

The following table combines low-frequency dynamic and static accelerations for these flight events, for satellites that comply with the frequency requirements in para. 7.8.1, and having their C G 850 mm above the separation plane. Longitudinal and lateral accelerations act simultaneously on the C G of the satellites.

FIG. 7.18 TEMPERATURE EVOLUTION DURING THE FLIGHT AT DIFFERENT LOCATIONS ON THE STRUCTURE
(Will be published in a subsequent issue)

FIG. 7.19 MASS / FREQUENCY DOMAIN FOR THE UPPER SATELLITE
(Will be published in a subsequent issue)

The following loads must be used for preliminary payload definition :

Flight limit loads applied to upper and lower satellite C G

Flight events \ Acceleration (g)	Longitudinal axis			Lateral axis
	static	dynamic	total	
Max. dynamic pressure	− 1.9	± 2.0	− 3.9 + 0.1	± 2.4
1st and 2nd-stage burn-out	− 4.7 0	± 3.2	− 7.9 + 3.2	± 1.2

- *negative longitudinal values indicate compression*
- *lateral loads may be applied in any direction*

These accelerations are applied to the greater part of the payloads, but certain areas are subjected to higher acceleration values.

Distribution of accelerations inside the satellites depends on their configuration. It is requested that a preliminary dynamic analysis be made at this stage. Dimensioning should take account of the safety factors defined by the payload authority (the Ariane authority calls for a minimum value at rupture of 1.25).

7.8.3. Dimensioning of secondary structures and flexible elements using the dynamic environment described in para. 7.7.1.3.

See para. 3.4.1.4.3.

7.8.4. Coupled analysis

See para. 3.4.1.4.4.

7.9. Qualification and mechanical acceptance of satellites

Each user is required to prove that his payload meets the dimensioning requirements in para. 7.8 above, by means of a qualification test and calculation files.

The elements required for drawing-up of the test plan are defined below, for each satellite. This test plan will be discussed with the Ariane authority. Tests are executed at individual satellite level, and no composite Sylda/satellite test is scheduled.

For execution of these tests, the payload authority can be provided with an adaptor and clampband. The strengh of the clampband/bolts assembly is limited to the loads corresponding to a bending moment of 48 000 Nm in the separation plane, if test bolts are used (with clampband tension 23 000 N), or 40 000 Nm if flight-type bolts are used.

7.9.1. Static qualification tests

See para. 3.4.1.5.1. The combined loads applicable are described in para. 7.8.2.

7.9.2. Sinusoidal vibration tests

7.9.2.1. Test levels

It is recommended that a factor of 1.5 be taken for vibration qualification tests.

Qualification and acceptance levels, applied at the Sylda/satellite separation plane, are given in the following table, and are recapitulated in figure 7.20 in the case of the qualification levels.

They are applicable individually to the upper and lower satellites. Any satellite must be capable of occupying either position.

	Frequency range (Hz)	Qualification level (0 - peak)	Acceptance level (0 - peak)
Longitudinal	5 - 10 10 - 18 18 - 28 28 - 100	5.6 mm 2.25 g 3.45 g 2.25 g	3.7 mm 1.5 g 2.3 g 1.5 g
Lateral	5 - 8 8 - 10 10 - 15 15 - 100	11.6 mm 3.0 g 2.25 g 1.5 g	7.7 mm 2.0 g 1.5 g 1.0 g

It is recommended that each test be preceded by a low-level vibration test, in order to detect the resonant frequencies of the satellite, and check any notching procedures.

The recommended sweep rate is 2 octaves/minute for qualification, and 4 octaves/minute for acceptance.

7.9.2.2. Notching

See para. 3.4.1.5.3.

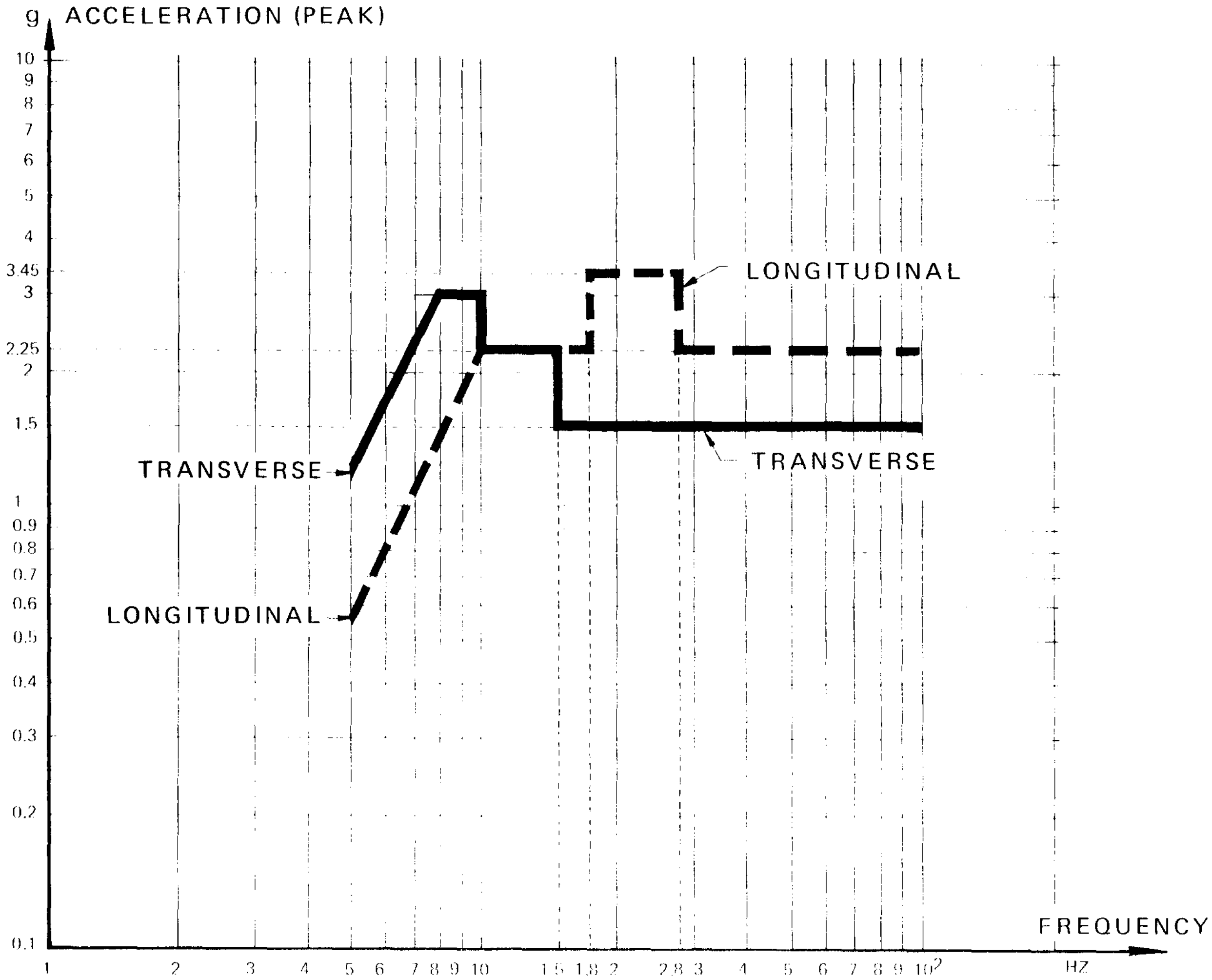


FIG. 7.20 SINUSOIDAL VIBRATION, QUALIFICATION LEVELS

7.9.3. Random vibration tests

See para. 3.4.1.5.4.

7.9.4. Acoustic vibration tests

See para. 3.4.1.5.5.

The choice of the spectrum depends on whether acoustic insulation is used or not (see para. 7.7.1.3.4.).

7.10 Electrical and radio interface

7.10.1. Continuity of Earth potential

See para. 3.5.1.

7.10.2. Standard umbilicals

The umbilical link for the upper satellite is via connectors P5 and P6. At the launch-vehicle end, the standard umbilical link has two 27-pin male connectors (Deutsch 79-27-0-PN (P6) and 0-PW (P5)), located in diametrically opposed positions in the separation plane, between the upper adaptor and upper satellite. The mechanical interface for these umbilical connectors, which are external to the lower frame of the satellite, is shown in figures 7.8 and 7.9. The cable is terminated by two umbilical connectors, P3 and P4, mounted on the Sylda structure, and a connection to the Electrical Umbilical Plug (Prise Ombilicale Electrique - POE) on the fairing, as in para. 3.5.2.2. Connectors P3 and P4 are disconnected automatically by the movement of the fairing, when the latter is jettisoned in flight.

The lower satellite has an umbilical link via two connectors, P1 (O-PW) and P2 (O-PN), identical to that for the upper satellite.

The principle of connection to the VEB electrical umbilical plug is as in para. 3.5.2.1.

The wiring of the Sylda adaptor also provides for connecting the wires in P1 and P2 used for transmission of satellite measurements during launch-vehicle flight to the VEB interface plugs.

Figure 7.21 shows the principle of umbilical connections. Umbilical connection availability is shown in figure 3.38 for the lower satellite (VEB/umbilical-mast links), figure 3.40 for the upper satellite (fairing/umbilical-mast links), and figure 3.41 for the umbilical mast junction box/launch centre links.

The Ariane authority will supply the user systematically with the female parts for the umbilical connectors, to be mounted on the flight model of the satellite (Deutsch DBAS 70-27-O-SN and OSW).

Maximum load on insertion or extraction is 180 N per connector.

7.10.3 Satellite measurement data transmitted by the launch vehicle.

Microswitches placed inside the satellite distancing springs provide for telemetry detection of satellite separation.

In addition to satellite/launch vehicle separation status reports, only three on/off signals ($<$0.5 V) are available for the two satellites.

7.10.4. Connection with vehicle checkout system

(" Banc de Contrôle Lanceur " - BCL)

See para. 3.5.7.

7.10.5. Radio transparency

7.10.5.1. Upper satellite : radio transparency of fairing

See para. 3.5.8., but note that antennae will not be located at the level of the radio-transparent boat-tail section, and therefore if radio-transparency is required, the possibility of placing radio-transparent ports in the area authorized for the location of access doors, or the use of passive repeaters, can be considered.

7.10.5.2. Lower satellite : radio-transparency of Sylda and fairing

The materials used in the construction of the Sylda are not radio-transparent. For satellites requiring radio-transparency, a system of transfer antennae can be used. There is a pick-up antenna inside Sylda, which drives a number of antennae on the outside of the Sylda adaptor, via a multiple feed. System gain in the 136-150-MHz band will be approximately – 25 dB.

The transfer system radiates through the radio-transparent boat-tail section of the fairing (see para. 3.5.6.). The arrangement may be finalized or modified by means of a 1/5-scale mock-up of the satellite, Sylda and fairing.

7.10.6. Radio compatibility

See para. 3.5.10., but note that in assessing potential interference from lower-satellite radiation, the power levels to be considered are those after transit through the Sylda transfer-antenna system.

7.10.7. Electromagnetic compatibility

See para. 3.5.11.

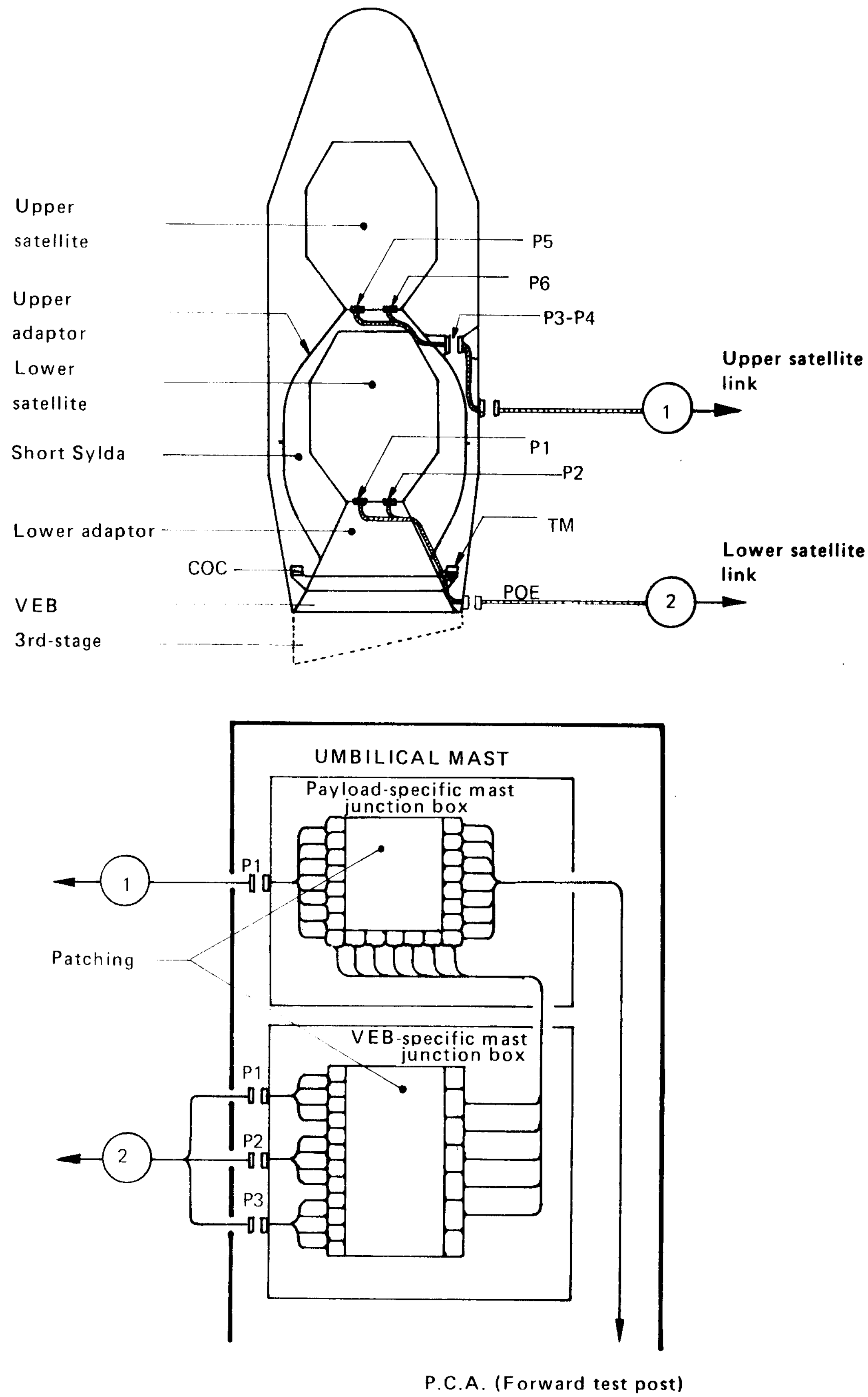


FIG. 7.21 UMBILICAL LINKS PRINCIPLE

7.11. Operations

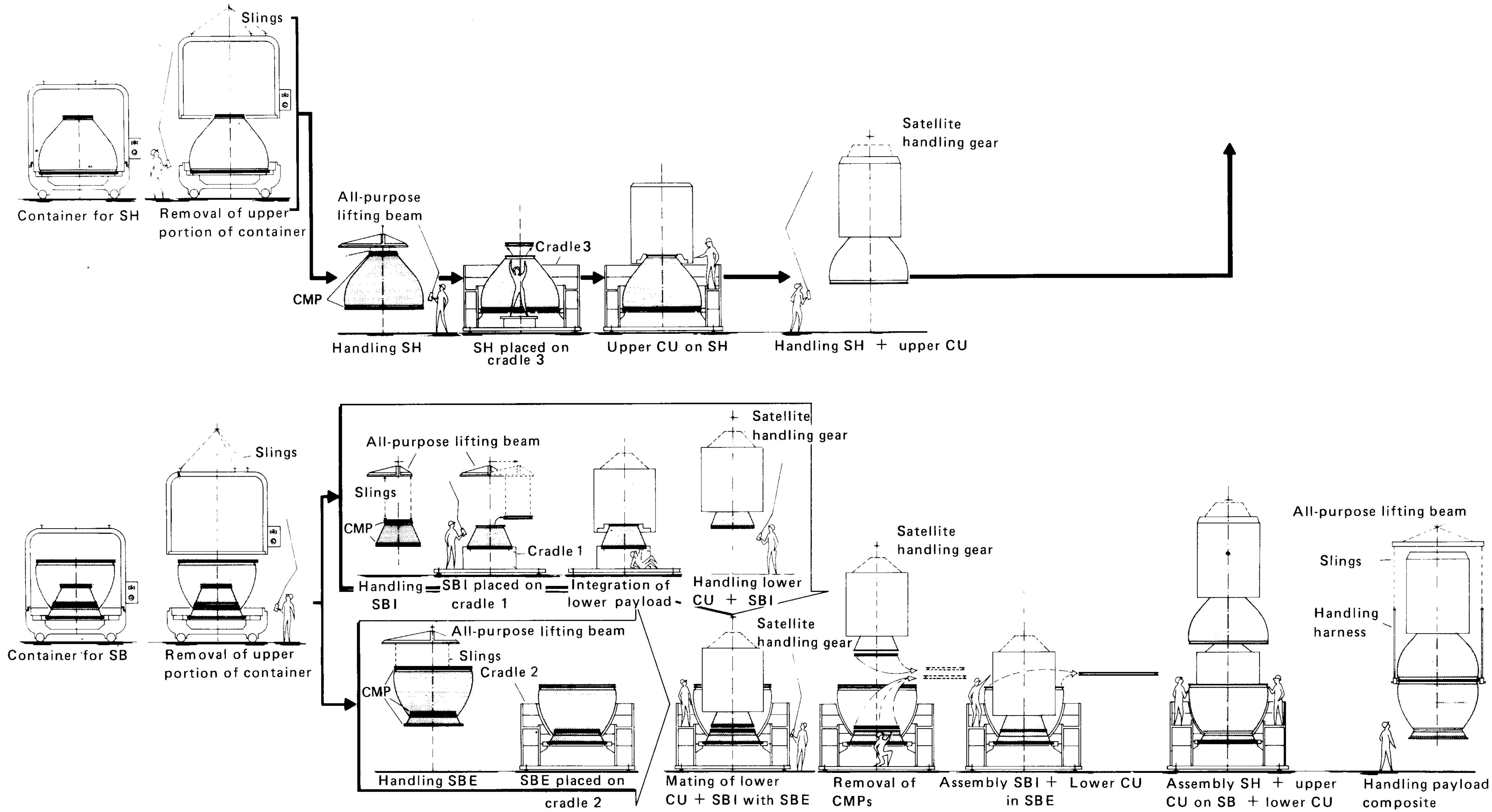
Operations are described in chapter 4. The following points are peculiar to a dual payload :

- After final inspection of each of the satellites (see para. 4.3.6.3.), the Sylda/upper-satellite/lower-satellite composite is assembled under the responsibility of the Ariane authority, in building S3 following the procedure illustrated in figure 7.22.

  Handling of each satellite with its adaptor is by means of the satellite-handling facilities. For this reason, the satellites and satellite-handling facilities must be capable of accepting an additional mass of 80 kg, corresponding to the adaptor.
- On completion of assembly, the dual payload is placed in the payload container (CCU), using handling facilities specific to Sylda.
- Operations on the launch-pad area, as described in para. 4.4., are carried out under the responsibility of the Ariane authority insofar as all operations on the dual payload composite, including assembly with the launch vehicle, are concerned ; and under the responsibility of each payload authority insofar as operations specific to each satellite are concerned.
- A typical time-schedule of dual-launch operation is (TBD).

7.12. Documentation

An individual DUA must be prepared by each payload authority, in accordance with the provision of chapter 6.



SH ("Section haute") = Upper section (of Sylda)

SB ("Section basse") = Lower section (of Sylda)

SBE ("Section basse externe") = Lower section – outside

SBI ("Section basse interne") = Lower section – inside

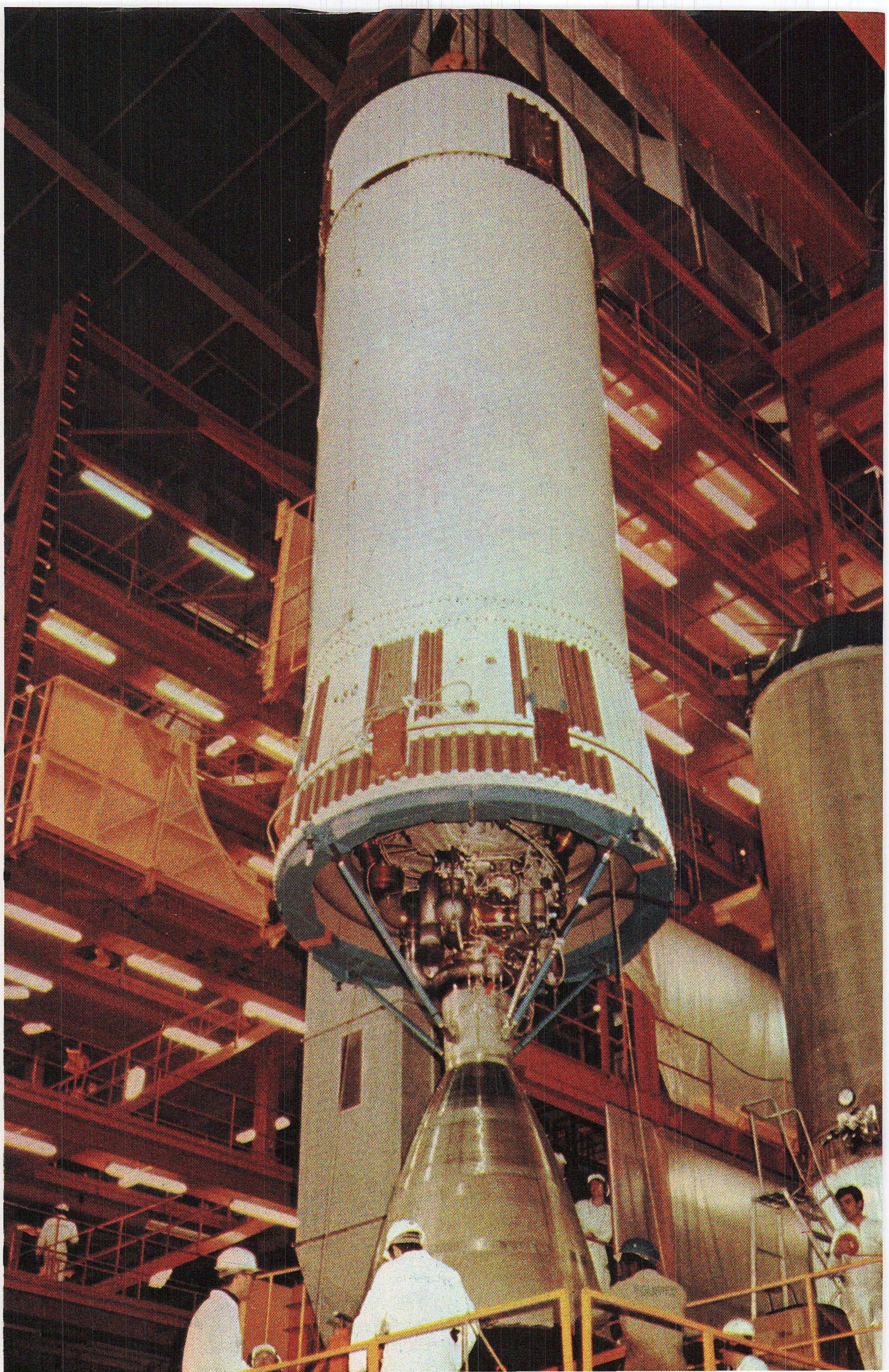
CMP ("Couronnes de Manutention et de protection") = Handling & Protection rings

CU ("Charge utile") = Payload

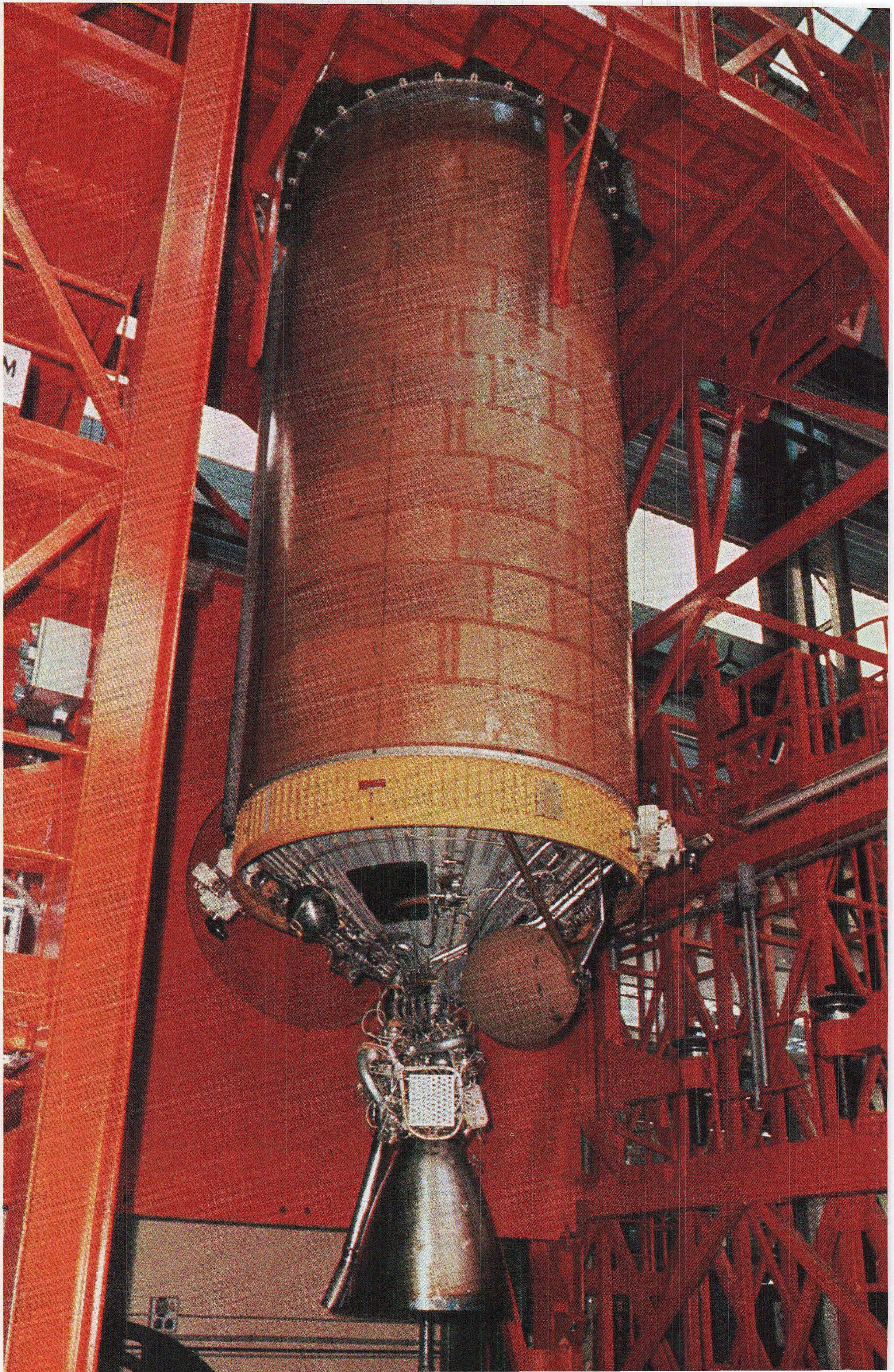
FIG. 7.22 INTEGRATION PLAN FOR DOUBLE PAYLOAD


esa

ARIANE first stage during integration
(SIL, Les Mureaux)



ARIANE 2nd stage during integration
(SIL, Les Mureaux)



ARIANE 3rd stage during integration
(SIL, Les Mureaux)

Description of launch vehicle — Annex 1

1. Introduction

The object of this annex is to provide an overview of the Ariane launch vehicle. The information given is designed to satisfy the initial curiosity of a reader wishing to have an approximate idea of the type of launch vehicle that he is considering using for a payload. The following description is therefore given by function and not by stage.

Ariane is a three-stage launch vehicle surmounted by an equipment bay and by a fairing which houses the payload. Its total lift-off mass is 210 tonnes, its height is 47.4 m, and its maximum diameter 3.8 m. Its performance enables a large variety of payloads to be placed into orbit, ranging from heavy 4 tonne satellites in low orbit to interplanetary probes towards Mars and Venus.

2. Electrical systems

2.1. Introduction

Most of the Ariane electrical systems are housed in the equipment bay at the top of the 3rd stage ; only a limited number of system elements are, for functional reasons, placed at various locations in the stages.

An on-board digital computer coordinates the activities of the various electrical subsystems of the vehicle. The electrical system provides the vehicle with total in-flight autonomy with the sole exception of the telecommand destruct signal sent from the launch centre.

Batteries are the only on-board electrical energy sources.

2.2. Guidance and control system (see fig. A 1.1)

The vehicle guidance system derives the correct vehicle attitude from the data of the inertial platform so as to optimize the payload mass to be placed in the desired orbit.

The vehicle control system generates the nozzle-swivelling commands for the various stages so that the vehicle will assume the attitude computed by the guidance system.

The guidance function is performed with the aid of the on-board computer flight programme, which executes the navigation calculations and implements the guidance law. It is active only from the flight of the 2nd

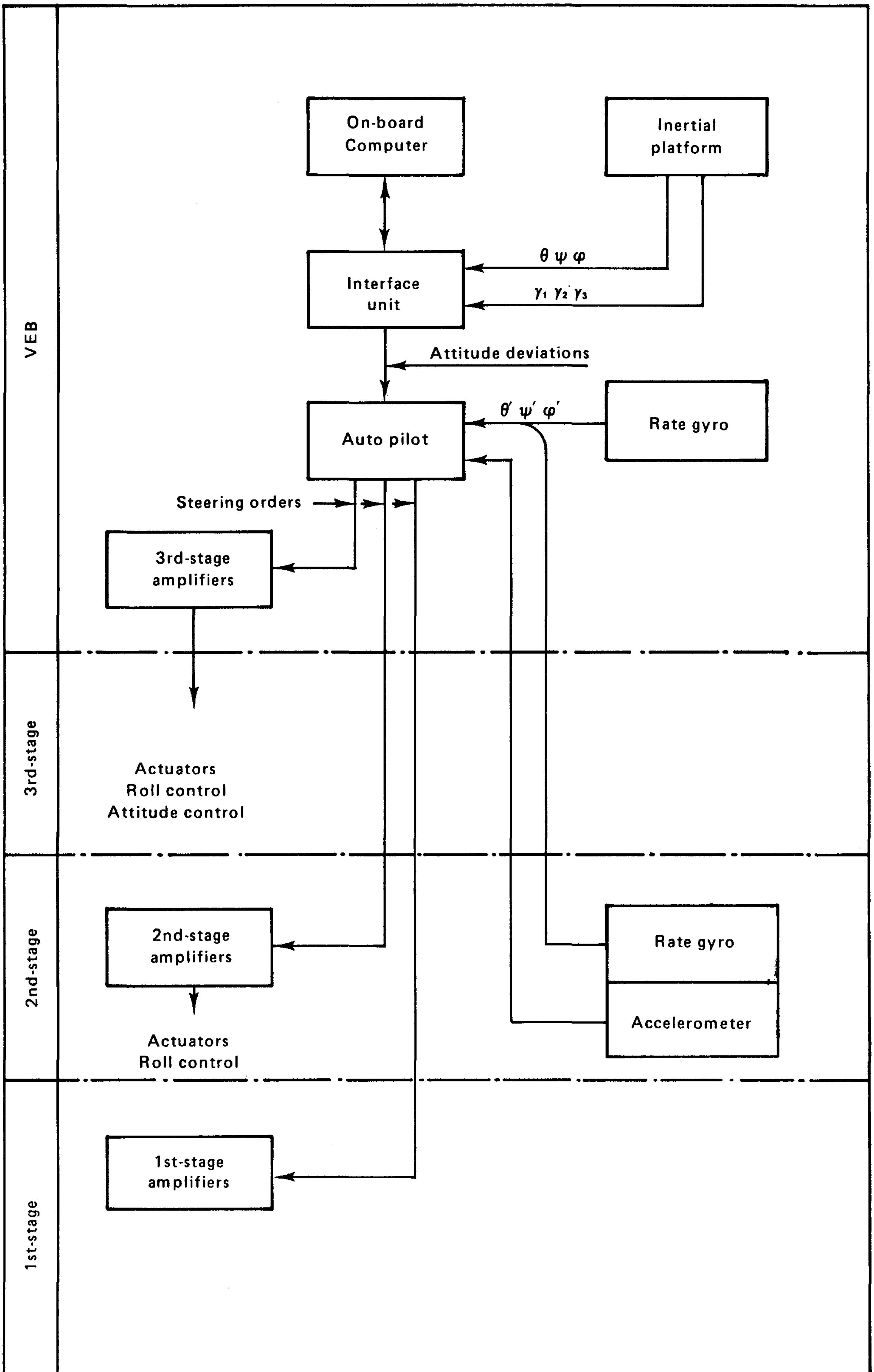


FIG. A 1–1 INERTIAL GUIDANCE CHAIN DIAGRAM

stage. 1st-stage flight is not guided, the vehicle following a pre-programmed attitude law.

The control function is performed with the aid of :

- sensors : inertial platform, rate gyros (and transverse accelerometers during a part of the 1st-stage flight);
- command-generating units :
  - the on-board computer ; this derives from the inertial platform the attitude deviation (difference between the actual attitude and the attitude required by the guidance system) and selects, depending on the flight phase, the gains, gyrometric rates, and vibration-mode filters for use in the autopilot unit ;
  - the autopilot unit ; this uses the attitude deviation supplied by the computer, the angular velocities supplied by the rate gyros and the transverse accelerations supplied by the accelerometers to determine the swivelling commands to be sent to the various stage nozzles ;
- executing units : each stage has amplification systems and servos commanding the position of the nozzles.

The 1st stage has four nozzles oscillating in the plane tangential to the thrust frame. This configuration enables pitch, yaw and roll control to be carried out by these nozzles alone.

The 2nd and 3rd stages each have a single nozzle and are equipped with ancillary systems for roll control.

These ancillary systems consist of sets of independent nozzles fed with hot gases from the gas generator on the 2nd stage and with cold pressurization gases on the 3rd stage.

The ancillary system on the 3rd stage is also designed to provide three-axis control after main engine cut-off. This enables the payload to be delivered accurately into the desired position and, if required, slow spin-up (10 rpm) to be provided.

2.3. Sequencing system (see fig. A 1-2)

This initiates all the commands needed for the execution and ordering of the various sequences governing the vehicle's in-flight behaviour. The sequencing system is commanded by the on-board digital computer, which transmits commands in the order prescribed by the flight programme.

The sequencing of commands is reset after the engine cut-off of each stage on detection of thrust decay.

The command transmitted by the computer through its interface unit is processed by a sequencing unit, which delivers directly signals matched to the executing units. Processing of the commands is done on two systems which are fully redundant from the interface unit to the executing units.

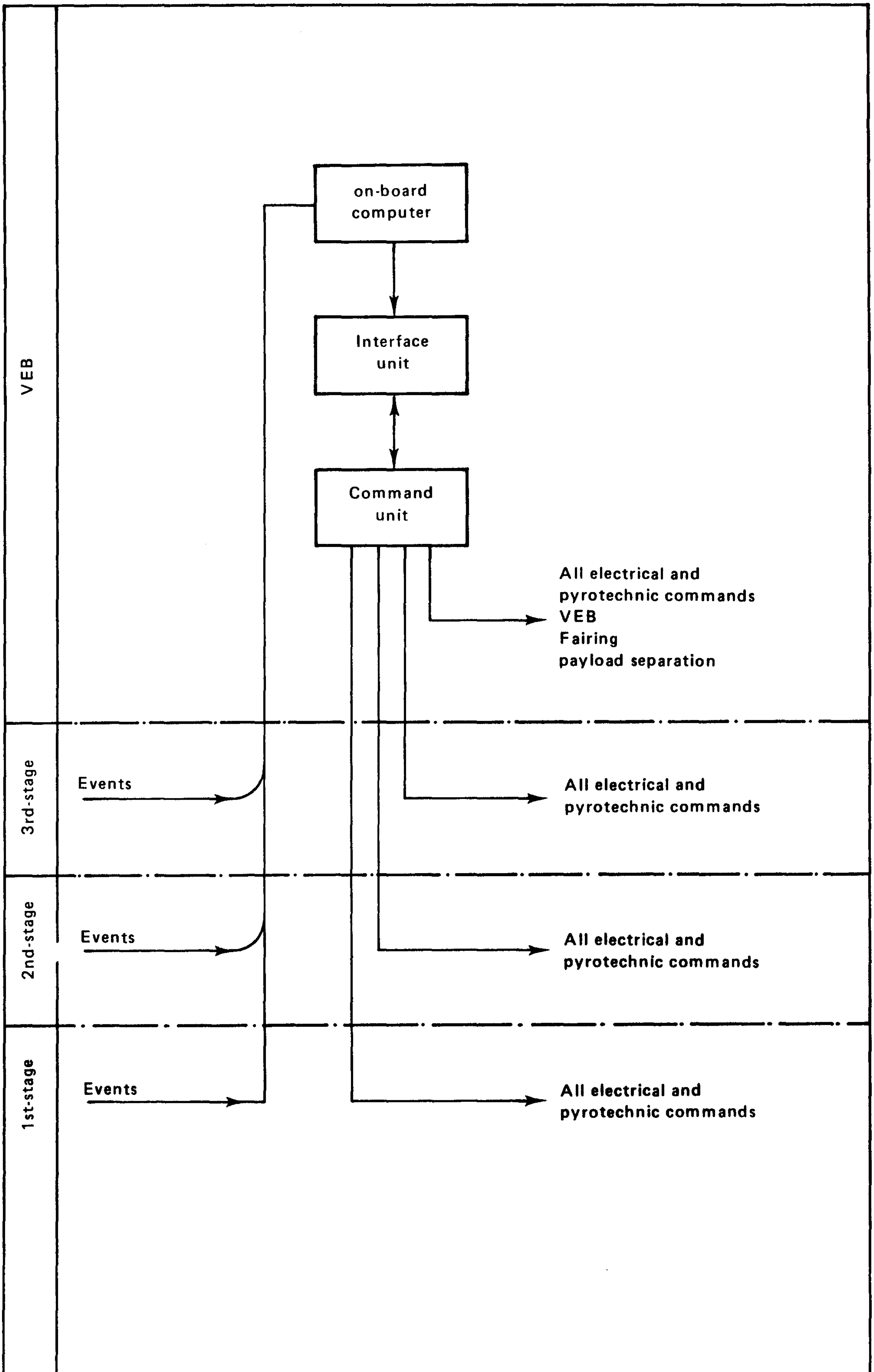


FIG. A 1–2 CONTROL LOGIC

The command signals from the sequencing system result in the following actions :

- electrical : change of PCM telemetry format, propulsion commands, operation of Pogo-correction systems ;
- pyrotechnic : acceleration retro-rockets, 3rd-stage engine ignition, separations, jettisoning.

2.4. Tracking and destruction system

This enables the vehicle to be destroyed in-flight in the event of abnormal behaviour that can constitute an environmental hazard. The destruct command signal is generated on board in the event of premature stage separation. The destruct signal commanded from the ground results from an operational decision based on processing of the data for the actual trajectory of the vehicle. These data are obtained in real time by the vehicle tracking system consisting of a tracking radar and transponders located in the vehicle equipment bay.

The tracking and destruction system is fully redundant, comprising two radar transponders and two telecommand receivers. The commands are generated in a safety unit, which decodes the telecommands. The 3rd-stage destruct charge is commanded directly by the safety unit. The destruct charges of the 1st and 2nd stages are each fired by a Self/Telecommanded Destruct logic circuit (" Destruction commandée et automatique " - DCA), which receives the destruct command ; the logic circuits of this system are located in their respective stages.

The telecommand-destruct receiver system may be inhibited by the receipt of the OFF command sent from the CSG while the vehicle is still visible or in range.

Details of the destruction devices are given in paragraph 5 below.

2.5. Telemetry system

The object of the telemetry system is to transmit to the ground the on-board measurements made during the flight of the vehicle, knowledge of which enables its behaviour and performances to be evaluated throughout the flight.

Two types of telemetry systems are used :

- a PCM system capable of transmitting a large number of slowly varying data ;
- an FM system capable of transmitting a small number of rapidly changing data (e.g. vibrations and shocks).

The operational version of the vehicle has only one PCM telemetry system for monitoring the complete vehicle. During the development phase, when other measurements are necessary for vehicle qualification, an extra PCM telemetry is installed for measurements on the 1st and 2nd stages and an FM system is installed in each stage.

3. Propulsion and pressurization systems

3.1. Introduction

The propulsion systems use the potential energy of the fuel stored on board to provide the vehicle with the kinetic energy required to place the payload in the desired orbit. The pressurization systems contribute to the proper operation of the propulsion systems by maintaining adequate pressure on the free surface of the various fuels in their tanks, thus keeping the hydrostatic pressure of the fluid circuits within limits compatible with correct operation of the various devices.

The 1st and 2nd-stage propulsion systems are similar and use the hypergolic liquid propellants UDMH and N_2O_4. The 3rd-stage propulsion system uses the cryogenic propellants liquid oxygen and liquid hydrogen, in a low-pressure chamber.

3.2. 1st-stage propulsion system

3.2.1. Principle

The system comprises four identical Viking engines which together develop a thrust of 2485 kN on the ground (2770 kN in vacuum), the specific impulse being 247.4 s (279.7 s in vacuum) ; the velocity increment imparted to the vehicle by the 1st stage is approximately 1800 m/s. Each engine forms an independent assembly supplied via individual valves from propellant and water tanks common to the four engines ; start-up is initiated simultaneously for the four engines by a ground command, ignition being spontaneous in the combustion chambers (hypergolic fuels). The operation of each engine is assured by slaving its core pressure to a reference " pilot " pressure supplied by nitrogen bottles. A gas generator fed with propellants in the stoichiometric ratio, and with water to lower the temperature, supplies both the gas turbine, which drives on a single shaft the propellant and water pumps, and the tank-pressurisation system. Lastly, a balance regulator compares the propellant-injection pressures and acts on the N_2O_4 pressure downstream from the pump in order to keep the propellant-mixture ratio constant in the combustion chamber.

3.2.2. Operation

During the automatic launch sequence, the vehicle checkout system verifies the correct operation of the vehicle, gives the signal for ignition of the four engines and monitors their behaviour. If everything proceeds normally, the command opening the vehicle release jaws is given three seconds after ignition. In the event of an incident, the release jaws are kept closed and the engines shut down.

The main operating characteristics are :

- Core pressure .... 55 Bar
- Turbopump rotor speed ....9 600 rpm
- H_20 delivery pressure .... 65 Bar
- UDMH delivery pressure .... 67 Bar
- N_20_4 delivery pressure .... 69 Bar
- Generator gas temperature .... 650 °C
- Generator gas pressure .... 35 Bar
- Pressurization pressure .... 5 Bar

Propulsion stops as soon as one of the fuels is depleted. The on-board computer detects this depletion by monitoring the acceleration of the vehicle and commands closure of the main fuel valves as soon as this becomes less than half its full-thrust value ; at the same time, it initiates the 1st/2nd-stage separation sequence described in paragraph 4.

3.3. 2nd-stage propulsion system

3.3.1. Principle

This system comprises a single Viking IV engine developing a thrust of 727 kN in vacuum, the specific impulse being 295.2 s. The velocity increment imparted to the vehicle by the 2nd stage is close to 2900 m/s.

The 2nd-stage propulsion system resembles that of the 1st stage except for the following differences :

- a single gimbal-mounted engine instead of four engines ;
- an independent roll-control system ;
- engine nozzle adapted to operation in vacuum ;
- helium-pressurization of fuels and water.

3.3.2. Operation

Engine ignition is commanded by the computer 0.3 seconds after 1st/2nd stage separation. Start-up is obtained by means of acceleration rockets (see paragraph 4) which provide the acceleration for settling the fuels after the 1st-stage engine cut-off. The engine nozzle, initially set in a zero position for pitch and yaw correction, receives swivelling commands as soon as thrust, and hence acceleration, attains its normal value. Guidance starts about 10 seconds after 1st-stage jettison.

The main operating characteristics that differ from those of the 1st stage are :

- Helium-bottle pressure .... 300 Bar
- Pressurization pressure .... 3,5 Bar
- Maximum roll torque ....1 000 m.N

Propulsion cut-off is effected, as for the 1st stage, by the computer, which simultaneously initiates 2nd/3rd-stage separation.

3.4. 3rd-stage propulsion system

3.4.1. Principle

This system comprises a single HM7 type engine developing a thrust of 61.8 kN in vacuum, the specific impulse being 440.6 s. The velocity increment imparted to the vehicle by the 3rd stage is approximately 5100 m/s. The engine is gimballed along the pitch and yaw axes. Roll control is carried out by a system of auxiliary nozzles fed with gaseous hydrogen.

Like the lower stages, the operation of the 3rd-stage propulsion system requires a turbopump fed with gases supplied by a generator. The latter is supplied with liquid oxygen and liquid hydrogen in a mixture ratio close to 0.9 in order to limit the temperature regime to 800°K.

The turbopump has two shafts. On the first, which rotates at about 60 000 rpm, are the turbine and the hydrogen pump and on the second, which rotates at about 13 000 rpm, is the oxygen pump ; the turbopump is actuated by a solid-cartridge starter and feeds the gas generator and the engine with liquid H_2 and liquid O_2. As these fuel are not hypergolic, the generator and the engine are ignited by pyrotechnic devices. Operation of the engine is controlled by a system for regulating injection flow rates into the generator by means of cavitating venturis and a pressure regulator on the oxygen circuit only. This device limits variations in engine operating parameters, particularly for the thrust and mixture ratio, whose variations are of the order of 1.5 %. The engine injector diffuses into the chamber liquid oxygen directly from the pump together with gaseous hydrogen at 150°K obtained by heating the liquid H_2 in the upstream (regenerative) section of the nozzle. A small part (6 %) of the liquid H_2 circulates in the divergent section of the nozzle, maintaining its temperature at about 1080°K ; it escapes at the extremity of the divergent section.

The oxygen tank is pressurized by means of helium, which is stored in a bottle at 100°K at 200 bar.

The hydrogen tank is pressurized by gaseous hydrogen at 100°K obtained by mixing liquid H_2 with gaseous hydrogen tapped at the exit of the regenerative section. This system is activated at engine start-up. During 1st and 2nd-stage flight, pressurization is effected with helium. In the event of self-pressurization due to considerable natural evaporation, the tank pressure is limited by a safety valve. The propellant feedlines are cooled throughout 1st and 2nd-stage flight by a continuous flow of a small quantity of waste propellant.

3.4.2. Operation

Engine start-up is commanded by the computer 3 seconds after 2nd/3rd stage separation. The stage nominal thrust is reached approximately 10 seconds later.

The main operating characteristics are :

- Core pressure .... 30 Bar
- Rotor speed of the turbine and the H_2 pump ....59 300 rpm
- Rotor speed of the 0_2 pump ....12 650 rpm
- Temperature of generator gases .... 880 °K
- Pressure of generator gases .... 24 Bar
- H_2 tank pressurization pressure .... 2.91 Bar
- O_2 tank pressurization pressure .... 2.11 Bar
- Maximum roll torque .... 800 N.m

The guidance system generates the propulsion cut-off command, which is sent by the computer. The propulsion cut-off sequence starts with the command which closes the 0_2 circuit valve at the gas generator input. For placing a 1700-kg payload in a 200/36 000-km geosynchronous transfer orbit, this command is given approximately 563 seconds after ignition.

4. Separation systems

4.1. Introduction

The operation of the vehicle involves the jettisoning of mass whose usefulness has ceased in the course of flight, e.g. the fairing above a certain altitude, and the distancing of a spent stage so that the next stage may be ignited.

4.2. Separation systems

The vehicle comprises four separation systems : two for the interstages, one for the fairing, and one for the payload. Each separation requires mechanical disconnection of the assemblies to be separated, followed by distancing. The systems used are similar for the interstage separation and different for the two others.

4.2.1. Interstage separation

This occurs at periods of virtually zero acceleration of the vehicle. Disconnection is obtained by a pyrotechnic cutting cord. The cord is fired simultaneously at two points and separation is achieved in less than 1 ms.

The forcing apart of the two stages is done by retro-rockets mounted on the lower stage, which impart an acceleration opposed to its velocity.

During the separation phase, acceleration rockets on the upper stage are fired and impart acceleration in the direction of its velocity ; this is designed to ensure that the fuels collect in the tank in such a manner that they flow properly to the engine ; it also contributes to forcing the two stages apart.

4.2.2. Fairing jettison

This occurs during 2nd-stage flight, that is when the vehicle is accelerating. The two half-fairings are separated from each other by pyrotechnic cutting along their contact generatrix and are ejected laterally, while remaining parallel to one another.

4.2.3. Vehicle / payload separation

This occurs after burn-out of the 3rd stage, which then remains attitude-controlled. Separation is achieved by ejecting the clampband by pyrotechnic action. The vehicle and payload are forced apart by the action of springs located around the periphery of the interface frame.

5. Destruction systems

The vehicle has three destruction systems, one for each stage.

The 1st-stage destruction system consists of pyrotechnic cords running along a generatrix of each tank.

The 2nd-stage destruction system is based on the same principle as the 1st stage, but has a single cord common to the two tanks (located near the common bulkhead, on a generatrix). The location of the 3rd-stage destruction system is identical to that of the 2nd stage, but a dihedral charge is used in order to allow for its distance from the wall structure (due to the presence of the Klegecell thermal insulation layer).

6. Structures

6.1. Introduction

All the structures are of light alloy with the exception of the 1st-stage water and propellant tanks, which are fabricated of steel in order to afford simultaneous protection from the temperature and corrosive action of the pressurization gases.

6.2. 1st-stage structure

The two propellant tanks are identical, cylindrical in shape (diameter 3800 mm, height 6700 mm) with ellipsoidal bulkheads. They are interconnected by means of a cylindrical inter-tank skirt of the same diameter as the tank and with a height of 2688 mm. The cylindrical part of the upper tank is prolonged by the forward skirt, which has the same diameter and a height of 1500 mm. The forward skirt supports the eight braking retro-rockets of the 1st stage and connects the latter with the interstage structure. The thrust frame is generally cylindrical in shape (a diameter of 3800 mm and height of 2300 mm) and has a caisson structure whose upper part connects with the UDMH tank and whose lower part is used for mounting the four engines. The toroidal water tank (average diameter 2700 mm, cross-section diameter 750 mm) is attached to the thrust frame structure by 16 rods. The engines, which protrude from

the vehicle, are shielded by cowlings ; four tail-fins, each with a surface area of 2 m^2, are used to improve aerodynamic stability during atmospheric flight.

6.3. 2nd-stage structure

The fuel tanks form a cylinder with hemispherical bulkheads (diameter 2600 mm, height 6515 mm) divided into two vessels by a common hemispherical bulkhead whose concavity faces forward. The pipework feeding the N_2O_4 (upper vessel) to the engine passes through the lower vessel (UDMH). The thrust frame consists of a stiffened skirt comprising a cylindrical section (with the same diameter as the tank and a height of 188 mm) and a frustum part (height 1350 mm) bearing the mounting flange of the gimbal unit. The aft skirt of the second stage is a frustum (height 1570 mm) and connects the 1/2 interstage frustum skirt (height 3310 mm) and the 2nd-stage thrust frame to which it is attached at the base of its cylindrical part. The rear-skirt/interstage-skirt frustum assembly transmits the 1st-stage thrust to the rest of the vehicle and provides continuity between the 1st-stage diameter (3800 mm) and the 2nd-stage diameter (2600 mm). The rear skirt carries the water torus (average diameter 2240 mm, cross-section diameter 340 mm) fixed by rods and the six acceleration rockets. It includes the pyrotechnic cutting system used for 1st/2nd-stage separation. The cylindrical-shaped forward skirt (height 1245 mm) prolongs the tankage and carries the braking retro-rockets. It connects the 2nd stage to the 2nd/3rd-interstage cylindrical skirt of the same diameter.

6.4. 3rd-stage structure

The fuel tanks form a cylinder with hemispherical bulkheads (diameter 2600 mm, height 6263 mm) divided into two vessels by an insulating intermediate bulkhead (two concentric spherical caps separated by a phenolic honeycomb layer under vacuum). The tanks carry thermal insulation over their entire surface.

The tank assembly is prolonged by a short forward skirt (2600 mm, height 451 mm) connecting with the equipment bay, and by a short rear skirt (diameter 2600 mm, height 288 mm) on which the thrust frame is mounted. The latter consists of a conical structure (height 1184 mm) together with a cylindrical section (diameter 2600 mm, height 499 mm) providing continuity with the rear skirt. The cylindrical section carries the propellant-settling rockets and the pyrotechnic cutting system used for 2nd/3rd-stage separation. The 2nd/3rd-interstage skirt (diameter 2600 mm, height 2730 mm) forms the connection between the 2nd and 3rd stages ; the lower part of this skirt is sealed off by an insulating disk for limiting heat exchange between the two stages.

6.5. **VEB structure**

With a total height of 1150 mm, this consists of a cylindrical section (diameter 2664 mm, height 300 mm) supporting a frustrum-shaped section, the upper frame of which is used for attaching the payload. The fairing rests on the upper frame of the cylindrical section. The frustrum section bears plates fixed by brackets at a height of 545 mm in relation to the base of the structure. The equipment-bay hardware is mounted on these honeycomb plates. The upper part of the frustrum section serves as an adaptor between the vehicle and its payload and comprises the vehicle/payload separation system. Heat exchange between the hydrogen tank, the equipment bay and the available volume under the fairing is limited by an insulating diaphragm.

6.6. **Fairing structure**

See paragraph 3.1.2.

7. Mass breakdown

The fully-equipped vehicle has a lift-off mass of 210 tonnes, including a payload mass of 1750 kg.

The mass breakdown is given in the following table :

	Lift-off mass (tonnes)			
	Structures	Propellants	Total	
			Item	Cumulative
Fairing	0.84		0.84	0.84
VEB	0.32		0.32	1.16
Third stage	1.2		9.45	10.61
Liquid oxygen		6.74		
Liquid hydrogen		1.55		
Second stage	3.8		37.85	48.46
UDMH		11.8		
N_2O_4		21.7		
Water		0.55		
First stage	13.3		159.65	208.11
UDMH		50.5		
N_2O_4		93.35		
Water		2.5		
Payload			1.75	209.86

Guiana Space Centre — Annex 2

I. Introduction

1.1. Situation

The Guiana coast is well situated for the location of a launching range.

Its proximity to the Equator, in an area outside the hurricane zone, the possibilities of launches due North and East over the ocean, and regular air and sea connections were factors which led to the choice of Kourou in French Guiana as the site for the CSG.

1.2. Tasks of the CSG

The Guiana Space Centre (CSG) was created and is managed by the Centre National d'Etudes Spatiales (CNES).

Within the CSG perimeter, ESA has established its launch facilities, as follows :

- Ariane Launch Site (" Ensemble de Lancement Ariane " - ELA), including the facilities required for the storage, final assembly, checkout and launch of Ariane.
- Payload preparation complex (" Ensemble de Préparation Charge-Utile " - EPCU), including the facilities made available to users for the preparation of their satellites.

Two agreements between the French Government and ESA define the rights and obligations of the two parties concerning the conditions for utilization of the ELA and CSG respectively. Under the terms of these agreements, the French Government guarantees ESA and its Member States free access to, and utilization of, the CSG facilities for their respective programmes. It also grants ESA priority for utilization of the CSG for its programmes.

ESA has set up permanent representation at Kourou (ESA Kourou Office), for the purpose of technical and financial control of ESA activities in Guiana.

The CSG's main tasks are as follows :

- Launch and injection into orbit of scientific and applications satellites ;
- Launch of sounding rockets or balloons, for scientific bodies ;
- Checkout and launch of any type of space vehicle.

These tasks comprise :

- The making available and operation of launch and measurement facilities, together with the associated material and operational support ;
- Restitution in real or deferred time of data concerning vehicle behaviour during launch.

2. French Guiana

2.1. Geography (see figs. A 2.1 and A 2.2)

French Guiana lies on the Atlantic coast of the Northern part of South America. It is an equatorial region covering about 90 000 square kilometers between latitude 2° and 6° North at longitude 50° West. It is bounded :

- to the North, by usually flat and marshy Atlantic coast covered with a recent alluvial deposit ;
- to the West, by the Maroni river, which forms the natural frontier with Surinam ;
- to the East, by the Oyapock river separating Guiana from Brazil ;
- to the South, by the frontier with Brazil, which is formed by the watershed of the Amazon basin.

Most of the territory is wholly covered with equatorial forest, only a coastal strip 15 to 30 km wide being habitable.

2.2. Climate

The climate is of the equatorial type, but is more moderate than its mean latitude would lead one to expect. With absolute extreme values of 19° and 35°, the temperature generally remains within a 25° — 30°C bracket, with a low daily variation.

Despite relatively high rainfall (annual mean 3m), there are two dry seasons : a short one in March with northeasterly trade winds, and the main dry season between August and December, with easterly trade winds.

The relative humidity is very high with a daily mean varying between 80 % and 90 %, corresponding to 18 to 22 g of water per cubic metre of air.

French Guiana lies outside the hurricane zone, and has light prevailing winds, mainly northeasterly. Despite its geographical situation, the coastal zone has a pleasant climate - breezy and sunny most of the year round.

2.3. Population

The current population of French Guiana is approximately 60 000, almost half of them in Cayenne. Most of the population is black or of mixed ethnic origin. This includes a sizeable proportion of Asians.

FIG. A 2—1 FRENCH GUIANA AND SOUTH AMERICA

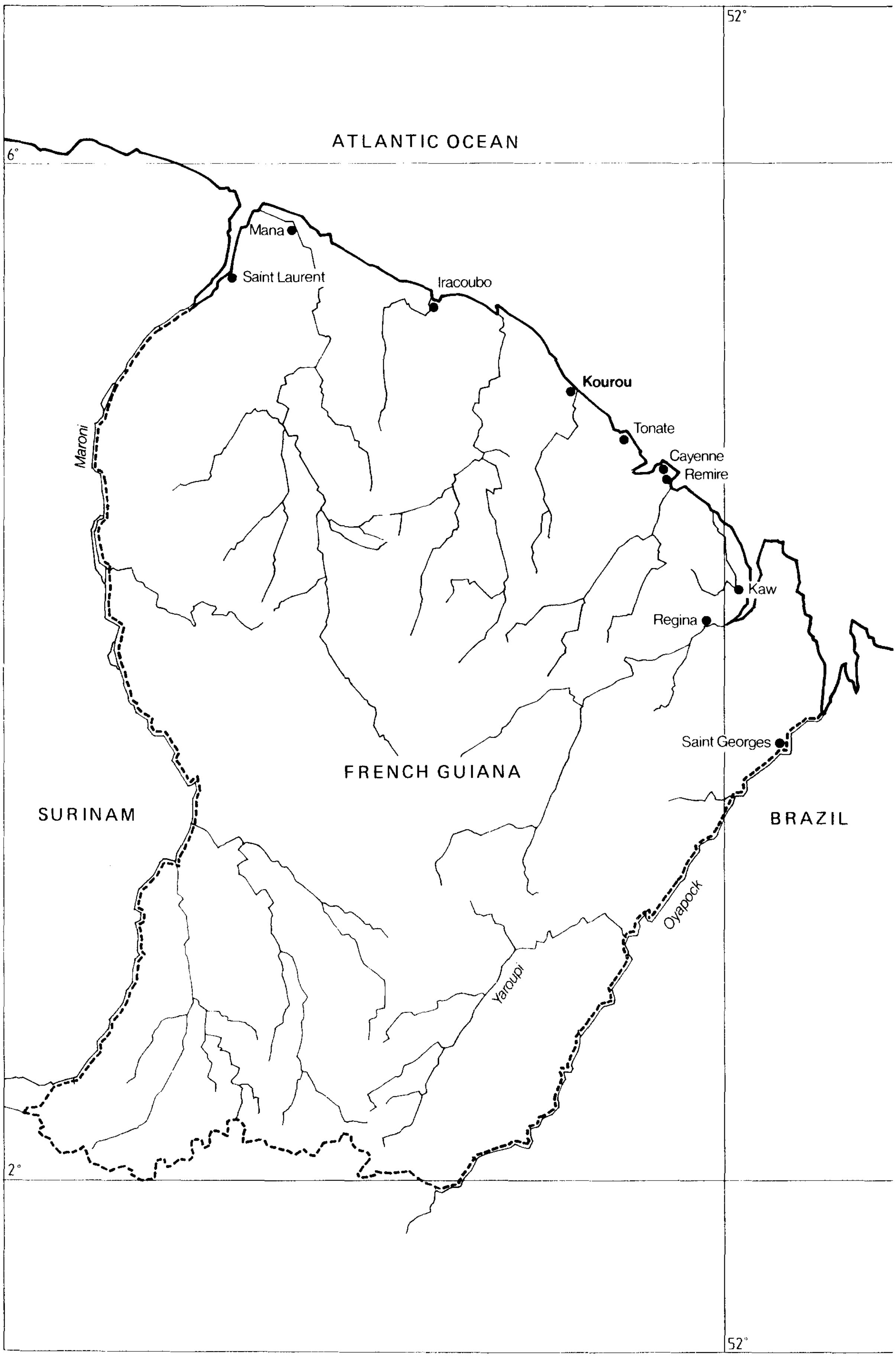


FIG. A2–2 FRENCH GUIANA

2.4. **Health**

Yellow-fever vaccination is mandatory for any visit to French Guiana exceeding 15 days.

Although many varieties of mosquito are present in Guiana, their bite is fortunately harmless.

Cases of dysentery and malaria are rare. Malaria is practically absent from the coastal strip (Kourou-Cayenne). Anti-malaria precautions are nevertheless recommended for persons entering the forest areas.

There are hospital facilities with very up-to-date equipment at Cayenne and Kourou.

2.5. **Infrastructure and communications**

- Sea links
  The port of Cayenne has three unloading areas :

  Cayenne proper
  Larvot } which can both handle cargo vessels of 1500 tonnes with a draft of 4 meters.
  Degrad des Cannes, to the South of the Cayenne peninsula, which can handle larger vessels of 6 meters draft.

  The port is linked to Kourou by road.

- Air links
  Rochambeau international airport at Cayenne has a 3200-metre runway, which enables wide-bodied jets to land in any weather. There are three or four flights a week from Paris, either direct or via the West Indies.

  These are also scheduled services between Cayenne and Brazil (Manaus and Belem), Guadeloupe, Guiana, Martinique, Peru (Lima) and Surinam.

- Road links
  A single road, the RN 1, links the main towns of Guiana (Cayenne - Tonate - Kourou - Sinnamary - Saint-Laurent du Maroni) and enables travellers to cross over to Surinam by the river Maroni ferry.

- Telephone - Telegraph - Telex
  Cayenne is linked by telephone and by telex to Paris, New York, Surinam and Fort de France.

All towns in Guiana are served by telephone and some by telex.

2.6. Kourou

The town of Kourou lies on the coast, approximately 65 km from Cayenne. The town was designed to cater for a population of 10 000, from the administrative, socio-economic, and basic-services points of view. Kourou has a current population of about 6000.

The town of Kourou has :

- a secondary school and three primary schools
- a day nursery and two infants' schools
- a medico-surgical centre and hospital
- a Youth Centre and Cultural Club
- a Church and Oecumenical Centre
- outdoor sports facilities such as a Sailing Club, Tennis Club, and childrens' playgrounds
- other facilities such as a swimming pool, a stadium and a sports ground
- a hotel complex including two high-class hotels with a total of 140 rooms (280 beds) and two self-service restaurants with seating capacity for 600.

Kourou town is supplied with electricity by a power station producing a total of 12 000 kVA. This power is distributed throughout the town and the Space Centre by two high-tension lines. The low-tension mains is the standard 220 - 380 V at 50 Hz. The Kourou power station is linked to the one in Cayenne.

The town is supplied with drinking water by a pumping station 32 km upstream from the estuary of the Kourou river.

Kourou has all the essential amenities of a modern town such as shops, supermarkets, restaurants, and craft trades of conventional and colourful kinds.

Abbreviations Annex 3

ABM	Apogee Boost Motor
ACM	Assistant Mission Director
ACU	Payload Adaptor
AZ	Azimuth
BCL	Launch Vehicle Checkout System
BNBD	Low-Level Bipolar Unbalanced
BRCU	Payload Interconnection Bay
CCU	Payload Container
CDL	Launch Centre
CM	Mission Director
CMP	Handling and Protection Ring
CNES	French National Space Studies Centre
COC	VEB Command Unit
COEL	Launch Site Manager
CPA	Ariane Project Manager
CPCU	Satellite Project Manager
CSEL	Launch Site Safety Officer
CSG	Guiana Space Centre
CST	Toulouse Space Centre
CU	Payload
DAM	Mission Analysis File
DCI	Interphase Control File
DDO	Launch Operations Director
DL	Launch Application
DUA	Application to Use Ariane
DV	Flight Director
ELA	Ariane Launch Site
EPCU	Payload Preparation Complex
ESA	European Space Agency
ESOC	European Space Operations Centre
FH	Radio-relay Link
GSOC	German Space Operations Centre
HNBD	High-Level Bipolar Unbalanced
KRU	Kourou
LAM	Measuring Instrument Laboratory
MCU	Payload Mass
MGSE	Mechanical Ground Support Equipment
MUA	Ariane User's Manual
OCOE	Overall Checkout Equipment
OL	Launch Order
ORS	Satellite Station Network Operator
PAM-D	Payload Assist Module - Class Delta
PCA	Forward Test Post
PCU	Payload Console

PF8	Ariane Servicing Tower - Platform N°8
PIP	Pyro Interception Plug
POC	Launch-Vehicle/Payload Combined Operations Plan
POCU	Payload Operations Plan
POE	Electrical Umbilical Plug
POS	Satellite Operations Plan
PPLS	Propellant and Pressurant Loading Systems
PR3	PR3 Storage Area
PSCU	Payload Safety Console
PY3	PY3 Storage Area
RAL	Launch Readiness Review
RAM	Mission Analysis Review
RAVL	Launch-Vehicle Flight Readiness Review
RAVS	Satellite Flight Readiness Review
RN	National Highway
ROO	Satellite Orbital Operations Manager
RPCU	Satellite Preparation Manager
RS	Safety Officer
S1	Building S1
S2	Building S2
S3	Building S3
S4	Building S4
SASY	Satellite/Sylda Separation System
SB	Sylda Bottom
SBE	Sylda Bottom, External
SBI	Sylda Bottom, Internal
SCAR	Attitude and Roll Control System
SCOE	Special Checkout Equipment
SH	Sylda Top
SYSY	Sylda Top / Bottom Separation System
TC	Telecommand
TM	Telemetry
UA	Acquisition Unit
ZA	Ariane Assembly Zone
ZL	Ariane Launch Zone
a	Semi-Major Axis
e	Eccentricity
i	Inclination
ω	Argument of Perigee
Ω	Ascending Node
Z_p	Perigee Altitude
Z_a	Apogee Altitude
M	Mean Anomaly
AZ	Azimuth
Ω_D	Descending node

This document has been published by :
— EUROSAT, 24 route des Acacias - CH - 1211 GENEVE 24
— With the assistance of RSCG - LEBLONG - MOINON - THIL Agency